NASA CR-143639

22296-6001-RU-000
1 OCTOBER 1974

FINAL REPORT

1

ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES AND RECOMMENDATIONS

(NASA-CR-143639) ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES. EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY (EOS) Final Report (TRW Systems Group) 112 p HC $5.25 CSCL 22B G3/18 N75-15685 Unclas 09222

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY (EOS)

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

RECEIVED FEB 1975 NASA STI FACILITY INPUT BRANCH

IN RESPONSE TO
CONTRACT NAS5-20519

TRW SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

22296-6001-RU-000
1 OCTOBER 1974

FINAL REPORT 1

ORBIT/LAUNCH VEHICLE TRADEOFF STUDIES AND RECOMMENDATIONS

EARTH OBSERVATORY SATELLITE SYSTEM DEFINITION STUDY (EOS)

PREPARED FOR

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GODDARD SPACE FLIGHT CENTER

IN RESPONSE TO
CONTRACT NAS5-20519

TRW
SYSTEMS GROUP

ONE SPACE PARK • REDONDO BEACH, CALIFORNIA 90278

CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

TABLES (Continued)

GLOSSARY

CDH	Communications and data handling
CDPF	Central data processing facility
EOS	Earth Observatory Satellite
EROS	Earth Resource Observation System
ETR	Eastern Test Range
FSS	Flight support system
HRPI	High resolution pointable imager
IFOV	Instantaneous field of view
KSC	Kennedy Space Center
LCGS	Low cost ground station(s)
LRM	Land resources management
MLI	Multilayer insulation
MMD	Mean mission duration
MODS	Multimegabit operational data
MSS	Multispectral scanner
MTTF	Mean time to failure
OMS	Orbital maneuvering system
PMMR	Passive microwave radiometer
SAR	Synthetic aperture radar
SEASAT	Sea Satellite
SEOS	Synchronous Earth Observatory Satellite
SMM	Solar maximum mission
SNR	Signal-to-noise ratio
SPMS	Special-purpose manipulator system
SSR	Scanning spectro radiometer
STS	Space transportation system
TM	Thematic mapper
TWT	Travelling wave tube
WBCS	Wideband communication system
WBDH	Wideband data handling
WTR	Western Test Range

1. INTRODUCTION

This report presents an evaluation of EOS design, performance, and cost factors which affect the choices of an orbit and a launch vehicle. Primary emphasis is given to low altitude (300 to 900 nautical miles) land resource management applications for which payload design factors are relatively well defined. Where appropriate, the treatment is extended to certain advanced missions (e.g., SEASAT and SEOS) with requirements which typify a broad class of future missions (e.g., non-sun-synchronous orbits and geostationary orbits).

The initial sections of this report present a mission model, orbit analysis and characterization, characteristics and capabilities of candidate conventional launch vehicles, and space Shuttle data. (The influence of launch vehicle and (primarily) orbit selection upon payload, spacecraft, and ground system design, and cost is then developed, with reference made to the more complete and general presentations of Reference 1.) The concluding section develops overall comparisons of the effect of orbit and launch vehicle selection upon system utility and cost for several specific cases.

The final selection of an orbit/launcher combination is a complete issue which extends beyond a simple comparison of Observatory and launch costs. The dimension of this process is enlarged considerably by questions of Shuttle application in retrieval and/or resupply on-orbit and the resultant effect on the cost of maintaining a specified operational capability over a defined mission cycle. Finally, such diverse scheduling factors as payload development cycles and Shuttle availability must be considered. The discussions and conclusions of the final section incorporate both the quantitative (weight/cost) data and these somewhat judgemental, but no less critical, system considerations. Recommendations can be summarized as follows:

---

(1) "Design/Cost Tradeoff Studies," TRW Report No. 22296-6001-RU-02, 15 July 1974.

1) EOS-A Mission. 336 nautical miles, sun-synchronous orbit; Thor-Delta 2910 launch vehicle.

2) Single Multispectral Scanner (MSS) Payload. 387 nautical miles, sun-synchronous orbit; Thor-Delta 2910 launch vehicle.

3) Single MSS plus Thematic Mapper (TM) Payload. 326 nautical miles, sun-synchronous orbit; Thor-Delta 2910 launch vehicle.

4) Dual MSS Payload. 326 nautical miles, sun-synchronous orbit; Thor-Delta 2910 launch vehicle.

5) Dual MSS plus TM Payload. 326 nautical miles, sun-synchronous orbit; Thor-Delta 3910 or Titan IIIB launch vehicle.

In cases 2) and 5) the spacecraft can be serviced on-orbit; others can be made serviceable with a launch vehicle upgrade (e.g., to 3910 or Titan IIIB). All cases are retrievable.

2. MISSION DATA BASE

Basic to selection of an orbit and a launch vehicle is a definition of the mission (or missions) to be considered and characterizations of the orbit and launch vehicle candidates.

2.1 MISSION MODEL

A general objective in spacecraft and subsystem development is flexibility of application with the land resources management (LRM) mission, a primary flight application. To this end, three categories have been dealt with in design:

1) The LRM Mission — implemented primarily with visible-spectrum instruments and sun-synchronous low-altitude orbits. Payloads are relatively well defined.*

2) Specified Advanced Missions — these missions, defined by the study contract, include: Synchronous Earth Observatory Satellite (SEOS), Solar Maximum Mission (SMM), and SEASAT. These missions distinctly differ from the LRM case. Payloads are partially defined.

3) General Future Missions — broad future mission classes have been hypothesized in subsystem design, to test the flexibility of the module designs.

Orbit and launch vehicle selection necessarily requires focussing on specific operational needs and desires. This report, therefore, concentrates on the LRM mission, but presents data which can be extrapolated to other missions.

The mission schedule is pivotal in the system decision-making process. Figure 2-1 is a mission schedule model received from NASA/GSFC on 14 June 1974, and modified somewhat during subsequent verbal discussions. It defines four basic missions, including the Earth Resource Observation System (EROS), and other related satellites which are candidates for modular implementation. The first mission (EROS) is of primary concern here, and has received emphasis in the report.

*The five-band MSS mission is included within the general category.

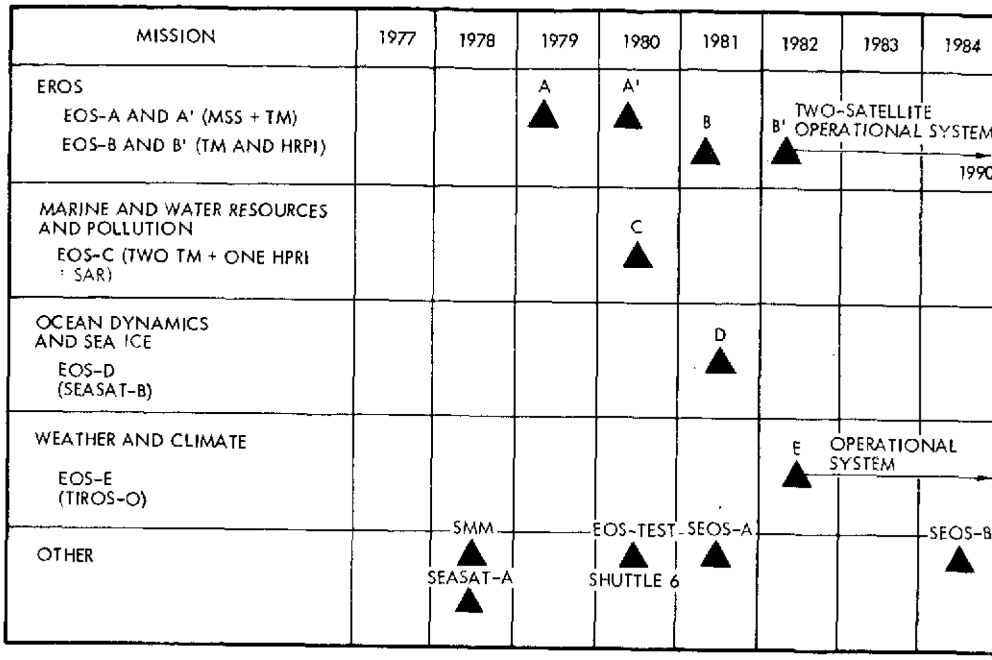


Figure 2-1. EOS Mission Model (June 1974)

As now defined, the Earth Observatory Satellite (EOS)-A and EOS-A' instruments are the five-band MSS and TM. According to Table 2-1, this payload will be succeeded by the TM plus high resolution pointable imager (HRPI) combination, which will later go operational. However, this model was augmented during discussions at GSFC on 22 June 1974 to include two candidate EROS operation/R&D missions:

1) Two MSS instruments giving adjacent swaths on a single satellite. Additional payload includes three wideband tape recorders (to support the MSS's, with one redundant); a wideband communication data handling system, and a TM (R and D payload). This satellite will be orbited to provide global coverage with a 9-day cycle (degrading to 17 to 18 days if one MSS is inoperative). The satellite will have a relatively high degree of redundancy.

2) Two satellites each with a single MSS launched so that the two-satellite system gives global coverage in 9 days (degrading to 17 to 18 days if one satellite is inoperative). Additional payload for each spacecraft includes two wideband tape recorders, a wideband communication data handling system, and a TM (R and D payload). Each identical satellite will be relatively nonredundant; redundancy being provided at the spacecraft level.

The choice of which mission to implement is affected significantly by the issues of this report, but also by areas beyond the scope of this document (e.g., mission cycle costs, Shuttle use optimization) that will be treated in Report 5.

Table 2-1. Advanced Mission Characteristics

Spacecraft	Mission Description	Typical Payload Instruments[1]	Payload Characteristics and Requirements[1], [2]			Data Rates	Attitude Control[2]		Orbit and Launch Vehicle
			Weight (lb)	Volume (cu. ft.)	Power (watts)		Orientation	Performance	
SEASAT	Demonstrate space monitoring of ocean surface conditions	• Synthetic aperture radar • Passive microwave radiometer • Infrared imager • Data collection system	500	600	500	0.5 kbit/sec to 10 Mbit/sec	Earth-pointing	0.25 deg accuracy	Low altitude, non-sun synchronous orbit (baseline is 391 n mi, 82 deg inclined); Thor-Delta launch for this payload
Solar Maximum Mission (SMM)	Study fundamental mechanism and effects of solar flares	• Ultraviolet magnetograph • EUV spectrometer • X-ray spectrometer • Hard X-ray imager • Low-energy polarimeter	1430	13.5	175	5 kbits/sec	Sun-pointing	5 arc-sec accuracy; 1 arc-sec drift in 5 minutes	300 n mi, 33 deg including orbit; Thor-Delta launch
Synchronous Earth Observatory Satellite (SEOS)	Resource and weather monitoring from stationary platform; timely warnings and alerts	• Large aperture survey telescope • Microwave sounder • Framing camera • Atmospheric sounder and radiometer	2640	350	145	60 Mbit/sec	Earth-pointing with scan	Point to 5 arc-sec (1σ); stability 1 arc-sec (1σ) in 12 minutes	24-hour geostationary; latitude and longitude stationkeeping; launched on Shuttle or large conventional launch vehicle with "Tug" stage

NOTES: (1) Data from the following reports:

- "SEASAT-A Phase I Study Report," W. E. Scull, NASA/GSFC, August 1973.
- "Solar Maximum Mission (SMM) Conceptual Study Report," NASA/GSFC Report X-703-74-42, January 1974.
- "Synchronous Earth Observation Satellite (SEOS)," NASA/GSFC Document, 1974.
- "Payload Characteristics for Gap Filler (5-band MSS) Mission," TRW Memo EOS-109, June 1974.

(2) Data from "SMM, SEASAT, ERS and SEOS Instrument Tables," NASA/GSFC, 1974.

As noted earlier, SMM, SEASAT, and SEOS have been considered in system design as potential applications of the modular spacecraft. Characteristics and potential payloads for these missions are summarized in Table 2-1.

2.2 ORBIT SELECTION CONSTRAINTS

Instruments for gathering earth resource data from space place stringent requirements on the orbit selection. These requirements will control such factors as the ground trace pattern, frequency of earth coverage, solar illumination angle, duration of pass time, and mission longevity. Review of earth observation mission requirements indicates primary interest in circular orbits ranging from 300 to 900 nautical miles and in geostationary operation.

For those missions desiring a minimum variation in illumination conditions at a given latitude, an orbital plane that rotates about the earth at the same angular rate as the mean angular velocity of the earth about the sun is required (sun-synchronous orbits). These orbits have the property of minimizing the variations in the angle between the sun and the local vertical at the subsatellite point (solar illumination angle) at a given latitude throughout the year. Variations of this angle will always occur, due to seasonal effects.

Table 2-2 summarizes the primary earth observation data user requirements. These suggest another important feature desired for low-altitude orbits, namely, repeatability of the ground trace over selected areas of the earth after a predetermined number of days. The repeating traces furnish a predictable pattern of coverage and permit direct comparison of similar data taken at regular intervals.

Each of the advanced missions specified have particular orbit requirements. SEOS requires a geostationary (24-hour) orbit to provide full coverage of CONUS with a short revisit period. SMM, a solar observation spacecraft, can be launched into a low orbit with an inclination selected to maximize the orbited payload.* SEASAT orbits are selected to give a good range of global coverage (i.e., high inclination)

*Alternatively, a high-inclination, sun-synchronous terminator (dawn-dusk) orbit can be selected to maximize sun viewing time.

and desirable swathing; however, illumination need not be maintained invariant, so that sun-synchronism need not be maintained.

Table 2-2. Earth Observation Data User's Requirements

APPLICATION DISCIPLINE	COVERAGE FREQUENCY	RESOLUTION	DATA TIMELINES
LAND PLANNING AND MANAGEMENT	WEEKLY TO ANNUALLY WITH PEAKS AT 18 DAY AND ANNUAL	PEAKS AT 15 AND 30 M	DAILY TO MONTHLY WITH PEAK AT MONTHLY
AGRICULTURE	WEEKLY TO 5 YEAR INTERVAL. STRONG PEAK AT 2 WEEK/ 18 DAY FREQUENCY	PEAKS AT 9, 15 AND SLIGHT PEAKS AT 30 AND 60 M	HOURLY TO MONTHLY
RANGE LAND	18 DAY TO ANNUAL WITH PEAKS AT 18 DAY AND ANNUAL	PEAKS AT 30 AND 60 M	DAILY TO WEEKLY
FORESTRY	18 DAY TO 5 YEAR INTERVAL WITH PEAKS AT 18 DAY AND ANNUAL	PEAKS AT 9, 15, 30 AND 90 M	DAILY TO MONTHLY
ENVIRONMENTAL MANAGEMENT	HOURLY TO ANNUAL WITH PEAK AT 18 DAY	PEAKS AT 15, 30 AND 90 M	DAILY TO MONTHLY
DISASTER WARNING, ASSESSMENT AND RELIEF	HOURLY TO 18 DAY WITH STRONG PEAKS AT HOURLY AND DAILY	PEAKS AT 15, 30 AND 60 M	HOURLY TO MONTHLY WITH PEAK AT HOURLY
WATER RESOURCES MANAGEMENT	HOURLY TO ANNUAL. SLIGHT PEAK AT 18 DAY	PEAKS AT 9, 15, 30 AND 90 M	DAILY TO WEEKLY WITH PEAK AT WEEKLY

The five-band MSS mission is a particular LRM case, subject to the same selection criteria and procedures used with other earth resource missions. The orbit analyses of Section 3.1 apply equally to this mission. Orbit selection for the other three special missions will not be considered further.

2.3 LAUNCH VEHICLE CANDIDATES

The following launch vehicles have been considered as candidates for the missions defined in Section 2.1:

- Thor-Delta 2910
- Thor-Delta 3910
- Titan IIIB
- Titan IIID
- Space Transportation System (STS) Shuttle.

These vehicles were selected for examination on the basis of their apparent cost effectivity for the candidate missions. Their characteristics are described in Sections 3.2 and 3.3.

The two-stage Thor-Delta 2910 is especially suited to low-earth orbit missions. Maximum performance is obtained using a Hohmann transfer flight mode. The first stage uses a fixed-thrust liquid propellant engine, with nine Castor II solid motors for thrust augmentation. The second stage uses a fixed-thrust bipropellant rocket with multiple restart capability; a nitrogen cold gas system provides pitch, yaw, and roll control during coast periods and roll control during powered flight. A cold gas retro system aids payload separation. Launch facilities for Thor-Delta exist at both the Western Test Range (WTR) and Eastern Test Range (ETR). More than 100 flights have been made to date.

The Thor-Delta 3910 is identical to the 2910 except the solid motors for first-stage thrust augmentation are Castor IV's, which provide increased payload capability. The 3910 is still undergoing development; its first flight, with a third stage (then designated as the 3914) is scheduled for December 1975 to place a communication satellite in geosynchronous orbit.

The Titan III vehicles were developed to meet the mission requirements of the U. S. Air Force. The Titan IIIB (SSB) consists of two bipropellant, fixed-thrust stages; the first, a stretched version of the standard core stage, and has been launched from WTR with an upper stage (which is not required for any of the currently defined EOS missions).

This vehicle has good capability for low-earth orbits; it is very efficient for elliptical orbits using a Hohmann transfer mode. However, an additional stage is required to circularize from the elliptical orbits; for the EOS missions; this stage is incorporated in the spacecraft.

The Titan IIID is a very energetic vehicle consisting of a standard core first stage, two 5-segment solid rocket motors for thrust augmentation, and the standard second stage. Like the IIIB, it operates most efficiently to low-earth elliptical orbits, requiring circularization to final orbit by the spacecraft. This vehicle has been launched from WTR.

The STS, also known as Shuttle, is a manned, two-stage bipropellant launch vehicle with solid rocket motor thrust augmentation. The bipropellant vehicles are reusable, and the upper stage ("Orbiter") is designed for landing like an airplane. The Orbiter will be equipped with mission-peculiar equipment capable of in-orbit servicing of EOS payloads, including retrieval and return to earth. Like the Titan class vehicles, the Shuttle has good capability to elliptical orbits and limited capability for circular orbits. Launch facilities to accommodate STS launches at WTR, for sun-synchronous orbits, are being planned.

3. ORBIT AND LAUNCH VEHICLE CHARACTERISTICS

3.1 ORBIT ANALYSIS

Factors governing selection of an earth resource mission orbit include ground trace pattern, frequency of earth coverage, solar illumination angle, ground station visibility, and launch vehicle performance. This section discusses the characteristics of each of the orbital parameters that influence the selection of an orbit for a low-altitude LRM mission.

3.1.1 Sun-Synchronous Orbital Characteristics

The line of nodes of an orbit will move westward or eastward in an inertial frame of reference depending upon whether the orbit inclination is direct (less than 90 degrees) or retrograde (greater than 90 degrees), respectively. This secular motion is primarily due to the J_2 term in the spherical harmonic expansion of the gravitational potential of the earth. A sun-synchronous orbit is one whose nodal rate of motion exactly matches in magnitude and direction the eastward motion of the mean sun. The appropriate inclination for such an orbit (always greater than 90 degrees) depends on the semi-major axis and eccentricity of the orbit. An approximate expression for the nodal rate is

$$\dot{\Omega} = - \frac{3J_2 R_E^2 \mu^{1/2}}{2a^{7/2} (1-e^2)^2} \cos i = - \frac{3}{2} J_2 \sqrt{\frac{\mu}{R_e^3}} \left(\frac{R_e}{a}\right)^{7/2} \frac{\cos i}{(1-e^2)^2}$$

where

$\dot{\Omega}$ = is the nodal rate (set to the mean solar angular rate of 0.9856 deg/day for sun-synchronous orbits)

J_2 = is the second harmonic in the earth potential function (1.08228×10^{-3})

R_E = is the mean radius of the earth (2.0925738×10^7 ft)

μ = is the gravitational constant of the earth (1.407648×10^{16} ft^3/sec^2)

a = is the orbit semi-major axis

e = is the orbital eccentricity

i = is the orbital inclination

The variation of inclination with circular altitude ($e=0$) for sun-synchronism is plotted in Figure 3-1.

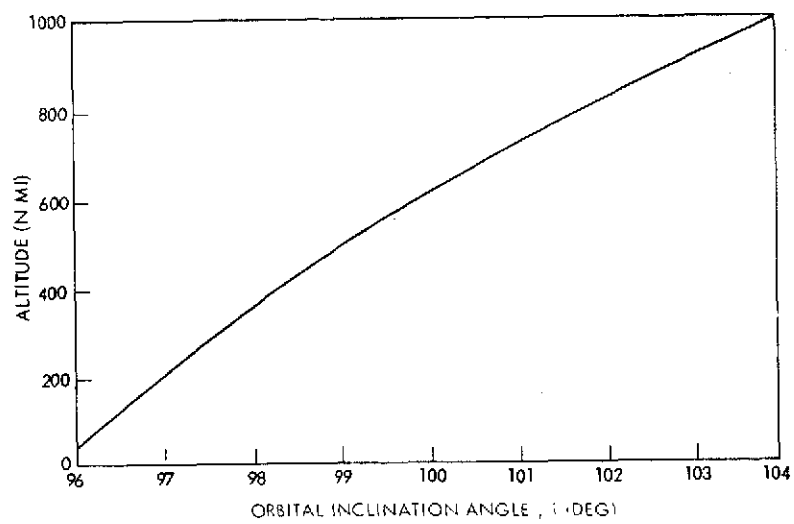


Figure 3-1. Sun-Synchronous Orbit Relationship Altitude Versus Inclination Angle

3.1.2 Earth Coverage and Swathing Considerations

Selection of the orbital altitude can have a profound effect on the utility of an earth observation mission, via the resulting ground track characteristics.

3.1.2.1 General Considerations

For earth resource missions, an orbit which produces a repeating trace pattern with full earth coverage (as constrained by orbit inclination) is desired. Such a characteristic can be attained by selecting from the discrete set of orbits such that there is an integral number (R) of satellite revolutions during the repeat cycle of N days; that is:

$$N = RP$$

Thus, the orbit period (P) must be a submultiple of the cycle period (N). Figure 3-2 defines the array of orbits giving repeating swath patterns for $N \leq 20$ days (Reference 2).

To prescribe the nature, extent, and frequency of coverage of features on the earth's surface, it is necessary to define the orbit trace pattern. A trace repetition parameter, Q, can be defined by the number of satellite revolutions which occur during one day. In general, Q is expressed as an integer plus a fraction n/d where n and d are relative prime integers. The denominator d is the whole number of rotations the earth must make relative to the orbit plane before the trace closes (i.e., d is equal to N). The numerator n determines the order in which the trace pattern develops. For circular orbits, the trace parameter Q is determined uniquely by the altitude. As shown in Figure 3-2, values of Q for the altitude range specified (300 to 900 nautical miles) are between 12 and 15 orbits per day.

The fundamental interval between two branches of the trace laid down on successive satellite revolutions is subdivided in d equal segments by d-1 subsequent crossings of the interval. Then the numerator n in the fractional part of Q is just the number of those segments between trace branches laid down on successive days. Hence, if n is either 1 or d-1, the pattern will develop steadily from west to east or from east to west, respectively; these are "minimum drift" orbits and lie near the zero drift lines of Figure 3-2 (denoted by the heavy dots). If $1 < n < d-1$, "skipping" occurs and more than one eastward or westward sweep (i.e., series of traces) is required prior to achieving trace repetition. The traces in such a sweep proceed from west to east if $n < d/2$; if $n > d/2$, the traces march westward across the fundamental interval.

For a circular sun-synchronous orbit, the only variables in the selection process are the orbital altitude and orbit plane orientation with respect to the sun (dictated by the desired solar illumination). From Figure 3-2, the selection of the trace parameter Q is limited to values of

(2) GSFC Note X-110-FO-456, "Swathing Patterns of Earth-Sensing Satellites and Their Control by Orbit Selection and Modification," J. C. King, December 1970.

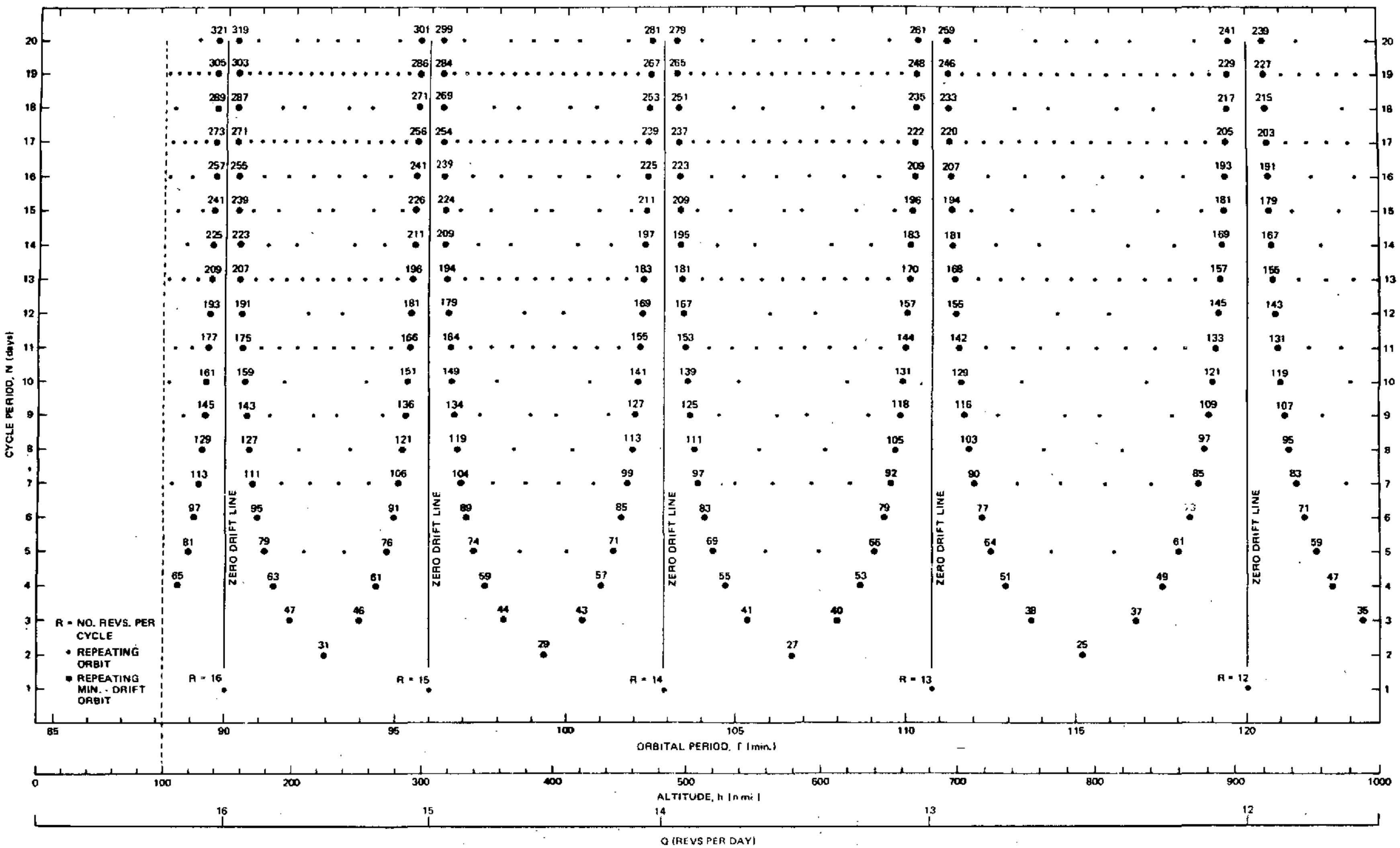


Figure 3-2. Array of Orbits That Produce Repeating Swath Patterns

between 12 and 15, the 300 to 900 nautical mile altitude range. An additional altitude constraint developed by desiring that the Shuttle be able to achieve the spacecraft circular orbit for servicing or retrieval, makes orbits with Q in the range 14 to 15 of primary interest.

The minimum number of revolutions, per trace repeat cycle, R, is constrained by the instrument swathwidth (W) and the desire to achieve full coverage with some fraction of overlap, η:

$$R > \frac{2\pi R_E \sin i}{W(1-\eta)}$$

Conversely, Figure 3-3 shows the minimum swathwidth required for full earth coverage for various repeat cycles in the allowable range of the trace parameter Q. The specified swathwidth for the thematic mapper is 100 nautical miles, for which the repeat cycle must be at least 16 days or greater with Q between 14 and 15.

Considering 16 and 17 day repeat cycles, Figure 3-4 indicates the relationship between altitude and the numerator of the fractional part of Q which determines the actual trace pattern. Figures 3-5 and 3-6 present the trace patterns of all values of n for a 17-day repeat cycle.

3.1.2.2 Swath Patterns for Pointable Instruments

An additional factor is significant in the selection of an altitude for a mission using a high-resolution pointable imager (HRPI) with a line-of-sight that can be offset from the nadir. In this case the range from which objects to be viewed can be selected is substantially greater than the 100-mile swathwidth of the thematic mapper (TM). Thus, with proper selection of the trace pattern, repeated viewing can occur in a significantly shorter time.* For example, consider Q equal 14 6/17 and a 30-degree HRPI offset as illustrated in Figure 3-7. In this case the interval between observations of some point can be reduced from 17 to 5 days. Table 3-1 presents this effect for N=17 patterns for both a 30- and 45-degree offset.

*Note that this is not the same as complete global coverage in a shorter period of time because for each region selected for viewing, many others are left unviewed.

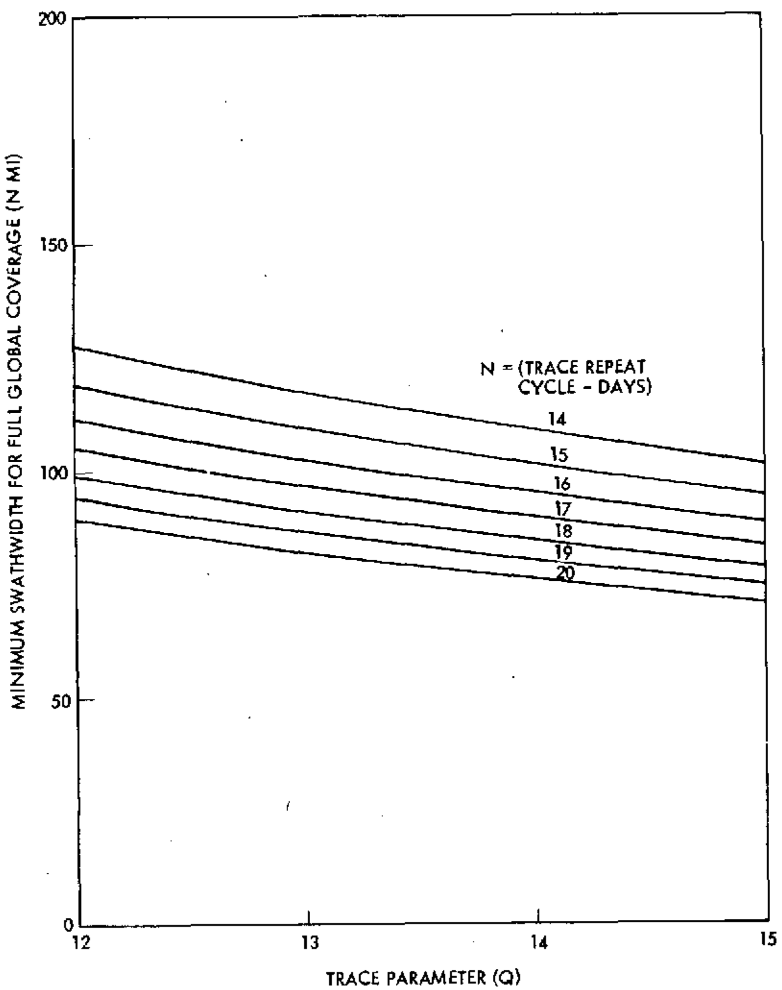


Figure 3-3. Swathwidth Requirements for Full-Earth Coverage

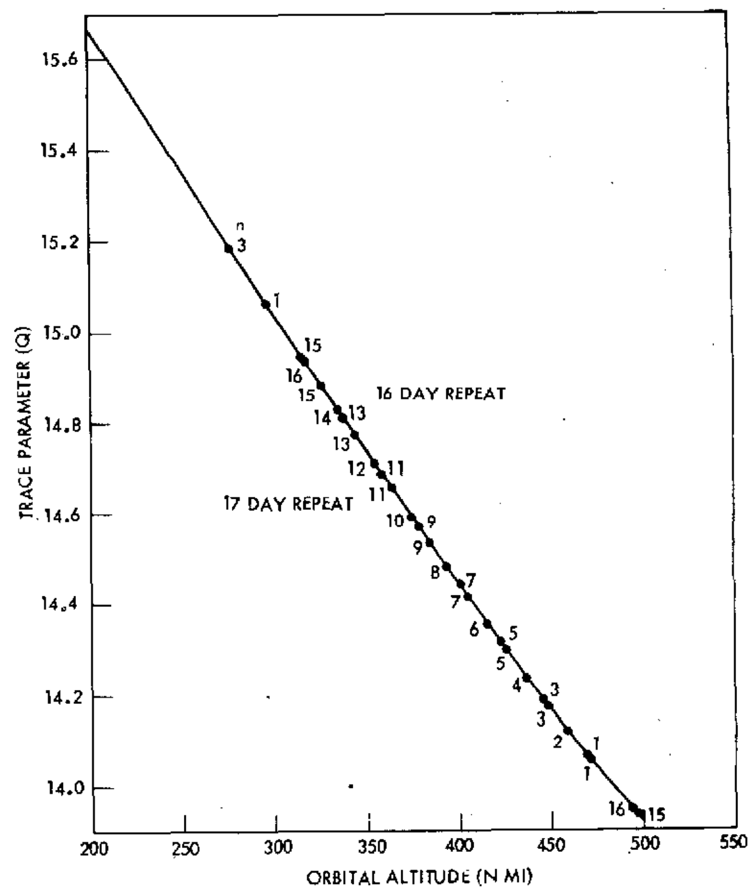


Figure 3-4. Tracing Parameter for 16- and 17-Repeat Cycles

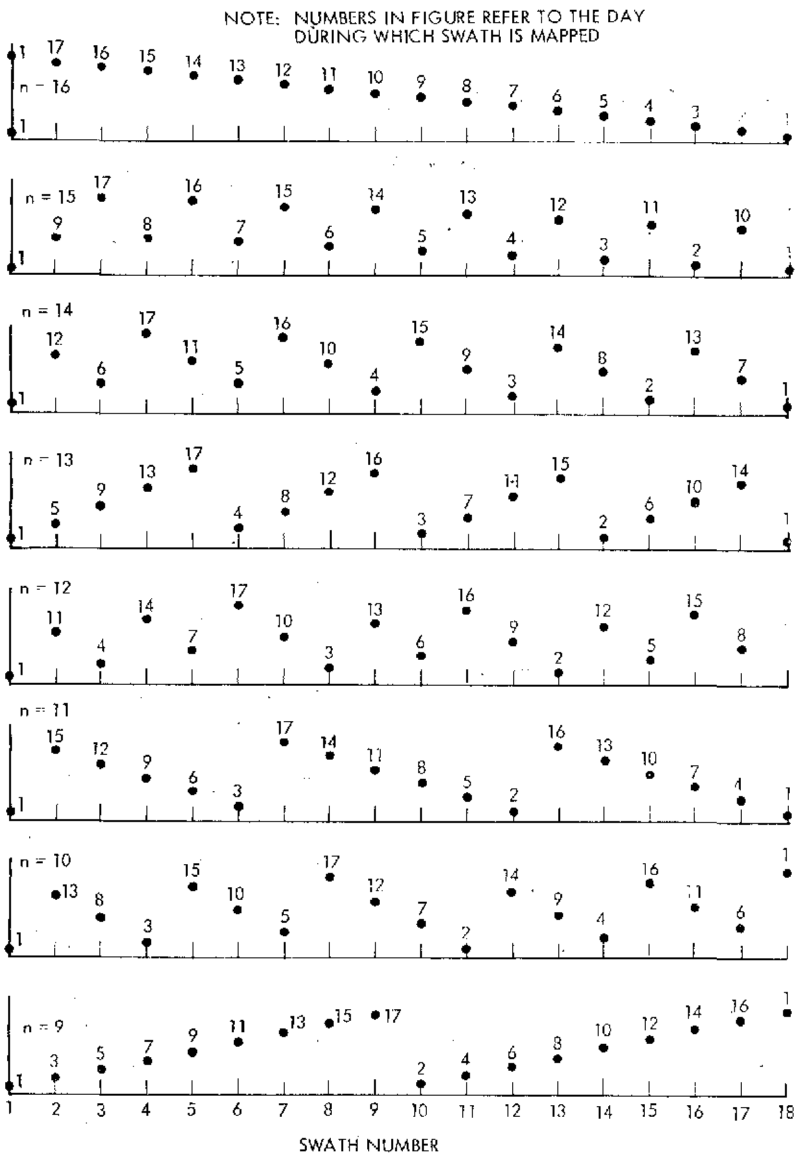


Figure 3-5. Swathing Patterns for 17-Day Cycle (n ≤ 8)

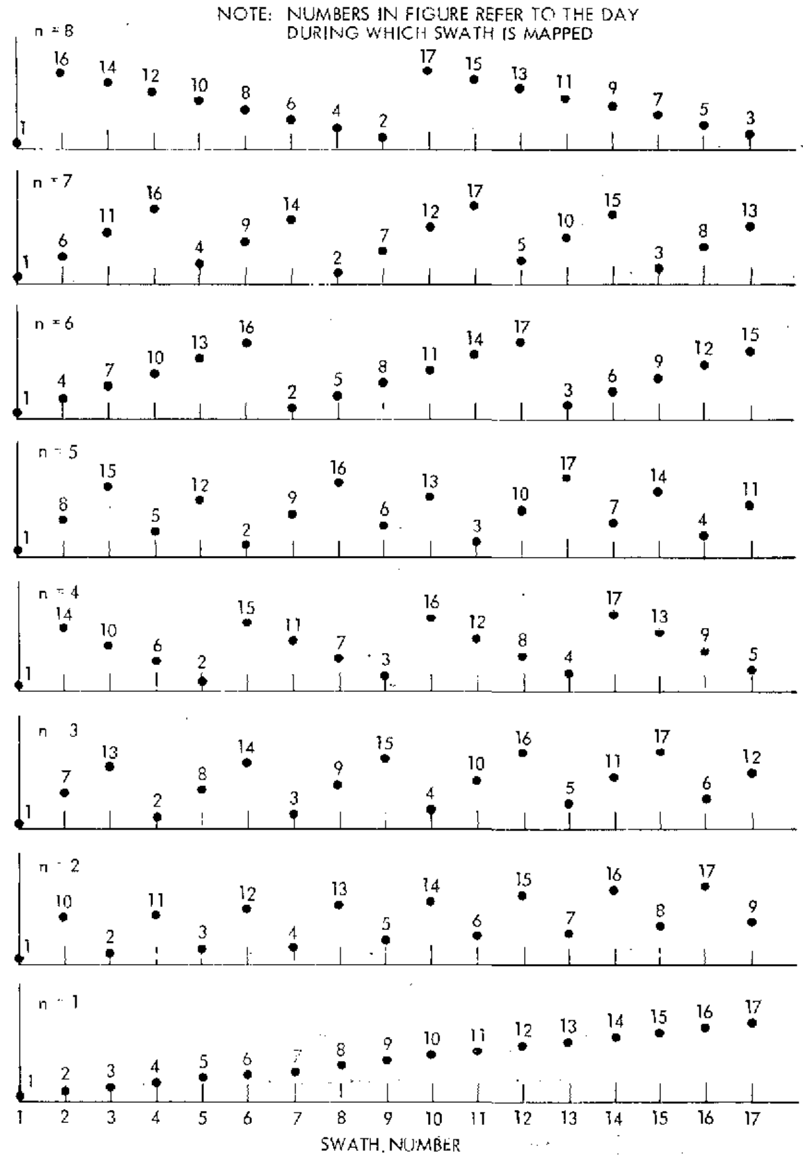


Figure 3-6. Swathing Patterns for 17-Day Cycle (n > 8)

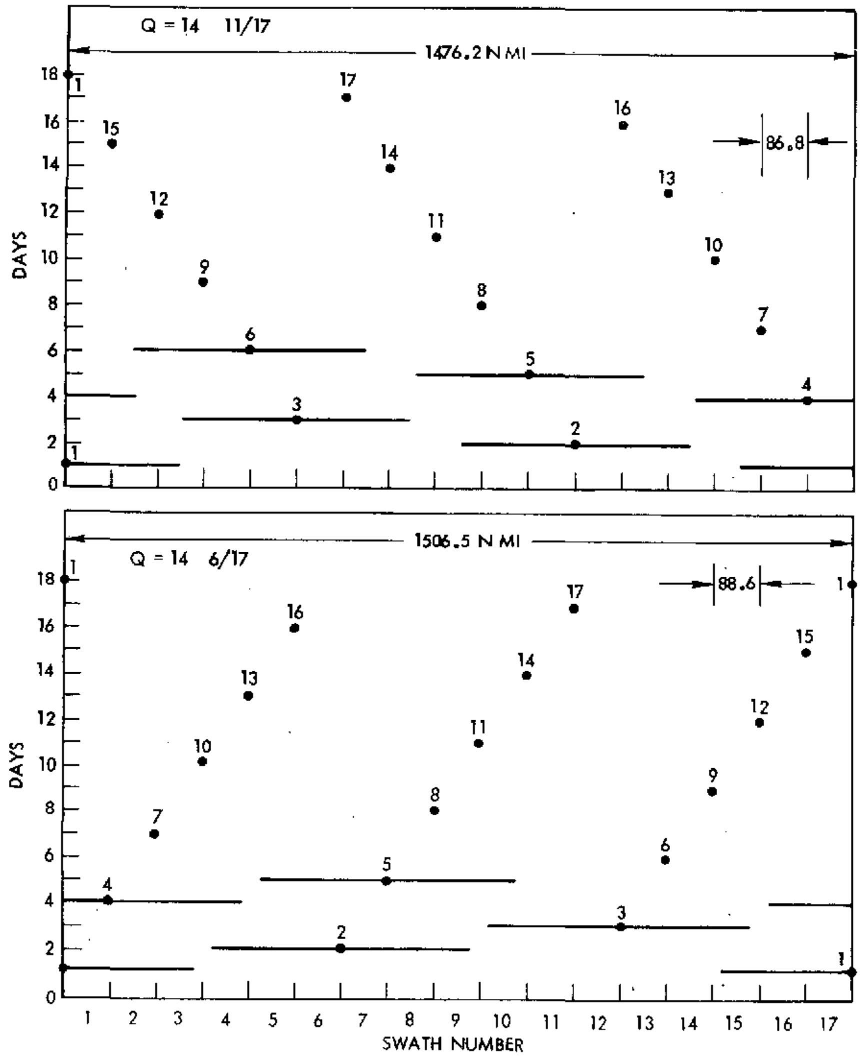


Figure 3-7. HRPI Offset Considerations

Table 3-1. HRPI Revisit Frequency for 17-Day Cycle Orbits

n	(30-degree offset)	(45-degree offset)	Altitude (n mi)
1	12	7	471.1
2	7	4	460.8
3	5	4	450.0
4	4	3	439.3
5	4	3	428.6
6	5	3	418.0
7	5	3	407.5
8	9	3	397.1
9	9	2	386.7
10	4	3	376.4
11	6	3	366.2
12	7	3	356.0
13	5	4	345.9
14	6	4	335.9
15	8	6	325.9
16	14	11	316.0

3.1.2.3 Swath Patterns With Tandem Instruments

An alternate approach to obtaining more frequent coverage is use of a wider swath to reduce the minimum number of days in a cycle. This can be accomplished with an instrument giving a larger swathwidth; or tandem instruments can be mounted on the same satellite. The latter approach can have significant advantages, particularly if the orbit is well selected.

Consider a payload consisting of two canted MSS-type instruments each having a swathwidth of approximately 100 nautical miles. In theory, an orbit can be selected such that the fully operational MSS payload will yield repeat coverage in about nine days and, in the event one instrument is inoperative, 17-day coverage using the remaining unit.

Figure 3-8 shows the trace pattern for an orbit meeting the above coverage objective. Table 3-2 summarizes the characteristics of a number of such orbits. Referring to Figure 3-2, these orbits all lie one increment in from the minimum drift orbits and, like them, are clustered in altitude ranges around the zero drift altitudes. In all cases the numerators (n) are either 2 or d-2. In addition the denominator (days in the total repeat cycle with a single instrument) must be odd to give the single MSS interlace coverage shown in Figure 3-8.

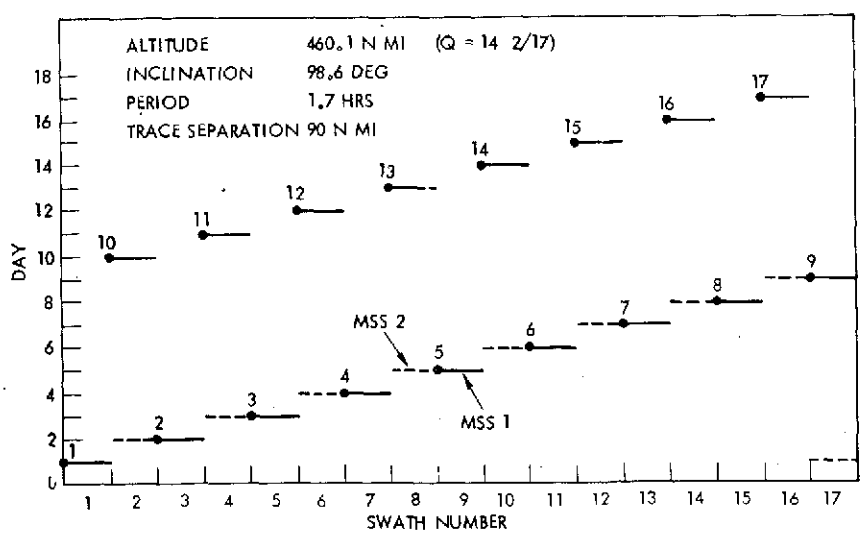


Figure 3-8. Typical Dual Instrument Orbital Trace Pattern

Table 3-2. Candidate Orbits for Tandem Instrument Payloads

Q	Altitude (n mi/km)	Days in Cycle	Revolutions in Cycle	Minimum Swath (n mi/km)*
15 2/17	286.6/531.1	17	257	84.2/156.0
14 15/17	325.9/603.9	17	253	85.5/158.4
14 2/17	460.8/853.9	17	240	90.1/167.0
13 15/17	504.8/935.4	17	236	91.6/169.7
15 2/15	284.1/526.4	15	227	95.3/176.6
14 13/15	328.5/608.7	15	223	97.0/179.7
14 2/15	457.9/848.5	15	212	102.1/189.2
13 13/15	507.8/941.0	15	208	104.0/192.7
15 2/13	280.7/520.1	13	197	109.8/203.5
14 11/13	332.0/615.2	13	193	112.1/207.7
14 2/13	454.1/841.4	13	184	117.6/217.9
13 11/13	511.7/948.2	13	180	120.2/222.7

*No overlap; 90-degree inclination.

Note that Table 3-2 considers only symmetrical-trace orbits, thus constraining altitude selection. It is apparent from Figures 3-5 and 3-6 that removing this symmetry constraint will allow a wider selection of orbital altitudes, which may be more in keeping with other orbit selection factors (e.g., launch vehicle capability and ground station coverage).

3.1.2.4 Multisatellite Systems

An alternative to a mission employing two instruments to double the available coverage is one in which two satellites are used, each with a single instrument. By placing these Observatories in the same orbit with properly phased orbit positions, the objectives noted in the previous section can be met. However, the swathing pattern is not so constrained as when the instruments are both on the same satellite, the only requirements being that one satellite (either one) give coverage in 17 days and the two together give complete coverage in 9 days.

Consider, for example, duplicating the n = 2 swathing pattern of Figure 3-8. If the satellites are phased so that the earth rotated through an odd number of swathwidths in the time between the two overflights, the two instrument swaths will provide the desired coverage.* Note, however, that the satellite should not be so close together (e.g., one swathwidth) that they appear in view of the ground station at the same time.

3.1.3 Orbit Phasing

Selection of orbit phasing (the equatorial crossing time) depends on illumination and, in a less clear manner, on cloud cover.

Illumination requirements favor near-noon orbits, with the peak illumination decreasing as the cosine of the offset from a noon equatorial crossing (e.g., 97 percent of peak for an 11 a.m. case; 71 percent of peak for a 9 a.m. case). Noon orbits, which provide overhead lighting and excessive reflections from bodies of water, are not desirable.

Cloud cover tempers phasing selection with payloads which operate in a visible spectrum. This factor tends to be a qualitative consideration; however, certain conclusions can be drawn. For example, morning orbits will be better than afternoon orbits, particularly during the warmer parts of the year. And of the morning orbits, those toward noon may be preferred during the winter and those away from noon more desirable in the summer.

Considering these factors, equatorial crossing times from 9 to 11 a.m. are appropriate. A baseline of 11 a.m. (descending) has been selected.

3.1.4 Ground Station Coverage

Ground station coverage can constrain the selection of an orbit, particularly in an operational mission where real-time coverage is desired using a limited array of ground terminals. Figure 3-9a illustrates this consideration for an operational scenario wherein 100 percent CONUS coverage is required via a single ground station at Sioux Falls, South Dakota. For altitudes significantly below the selected baseline of 386.6 nautical miles, coverage is lost in Florida, Maine, and on the Pacific Coast.

*The two cases are not quite equivalent. In the earlier one, instrument image overlap will be fixed by their mounting on the vehicle. In the dual satellite system, swath overlap will increase with latitude.

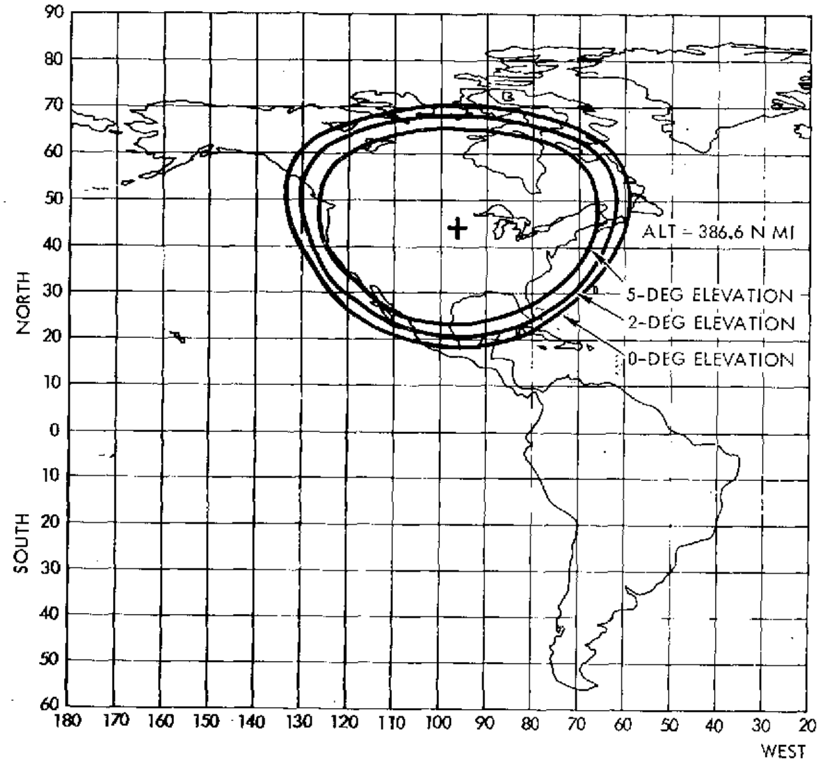


Figure 3-9 (a). Satellite Visibility from Sioux Falls for h = 386.6 Nautical Miles as a Function of Elevation Angle

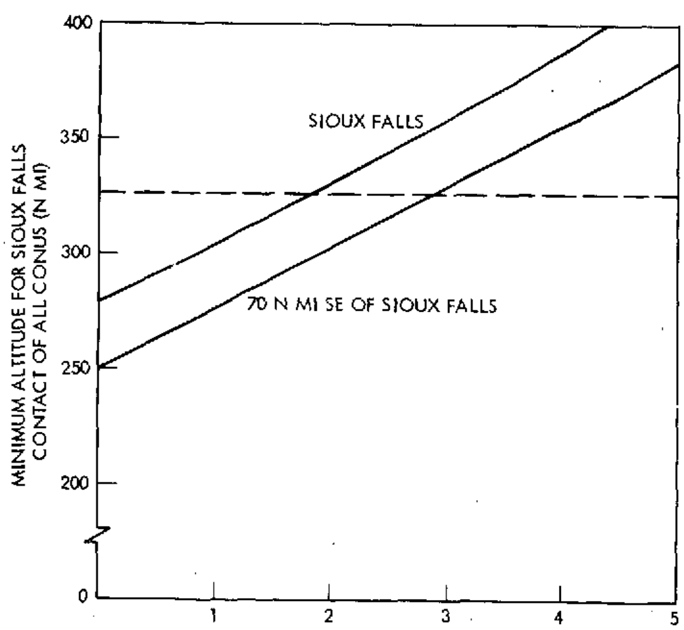


Figure 3-9 (b). Minimum Altitude for Single-Station CONUS Visibility as a Function of Spacecraft Elevation

Figure 3-9b shows the single station visibility situation in more detail, with satellite elevation above the horizon as a parameter. As indicated coverage is enhanced by locating the ground station about 70 nautical miles southeast of Sioux Falls. For a typical operational altitude, 325 nautical miles (see Section 7), 100 percent CONUS visibility can be achieved if operation at elevations as low as 2 degrees is feasible.

3.1.5 Orbit Maintenance

The most significant perturbation on low altitude orbits is atmospheric drag. For a spacecraft in orbit, the drag force will cause a deceleration of the body of magnitude.

$$\frac{dv}{dt} = \frac{1}{2} \rho v^2 \left(\frac{m}{C_d A}\right)^{-1}$$

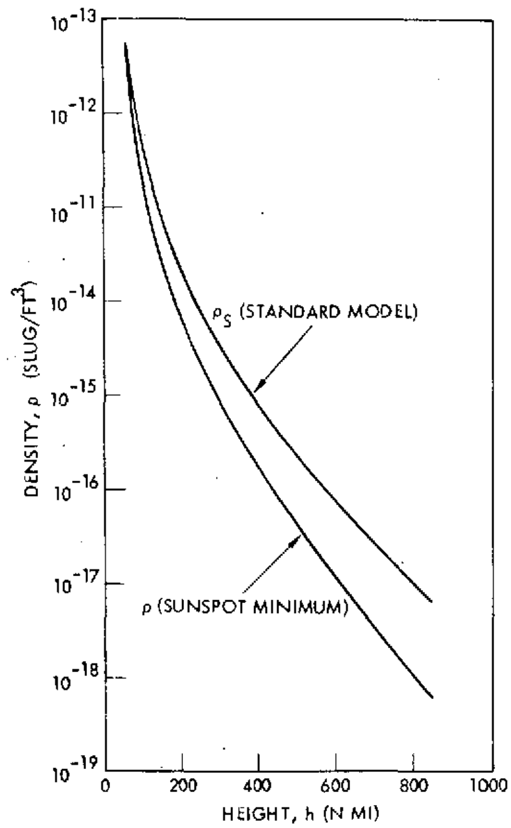


Figure 3-10. Atmospheric Density Model

In the above expression, the mass-to-drag ratio ($m/C_d A$) is a property of the spacecraft and its orientation with respect to the velocity vector. The term $q = \rho v^2/2$ is the dynamic pressure and depends on ρ, the atmospheric density, which is, in turn, a complex function of many variables (such as altitude, local time of day, and solar activity). Figure 3-10 gives typical values of density as a function of altitude for the combination of conditions leading to maximum and minimum densities, and provides a quantitative means of assessing the effect of drag on a satellite's orbit.

It can be shown that the change in semi-major axis for a circular orbit over one revolution due to atmospheric drag is approximated by

$$\Delta a = 2\pi \left(\frac{m}{C_d A}\right)^{-1} \rho\, a^2 \quad ft/rev$$

Since the semi-major axis determines the orbital period, and thus the ground pattern, orbital corrections are required to maintain the semi-major axis within some specified tolerance.

The swath advance error per day due to an error in the semi-major axis is given by:

$$\Delta S = \left(3\pi \frac{R_e}{a}\right) \Delta a$$

The above equations can then be appropriately combined to produce estimates of the time between velocity corrections to limit swath advance error to a specified value and the total velocity correction required per year. These data are presented generally in Figures 3-11 and 3-12 and specialized to Titan and Thor-Delta EOS configurations in Figure 3-13.

3.1.6 Orbit Achievement

For some orbital altitudes, it is either inefficient or impossible to use the Titan and Shuttle launch vehicles to place the EOS directly into the operational orbit. Thus, the spacecraft propulsion system is required to supply the impulse to achieve the circular operational orbit. The Titan and the Shuttle can place the EOS into an elliptical orbit with 100 nautical mile perigee altitude and apogee altitudes equivalent to proposed EOS circular operational altitudes. The orbit achievement maneuver by the spacecraft will then consist of a circularization burn at apogee of the injection orbit. The impulsive ΔV for circularization at various apogee altitudes is given in Figure 3-14. Note that the most efficient use of a Thor-Delta leads to direct injection into the operational orbit.

3.1.7 Geostationary Orbits

A satellite in circular orbit at 19,323 nautical miles altitude will circle the earth in exactly 24 hours. Direct injection into a geostationary orbit is not possible with any launch vehicle. Transfer from a 100 nautical mile circular parking orbit to a geostationary orbit generally requires

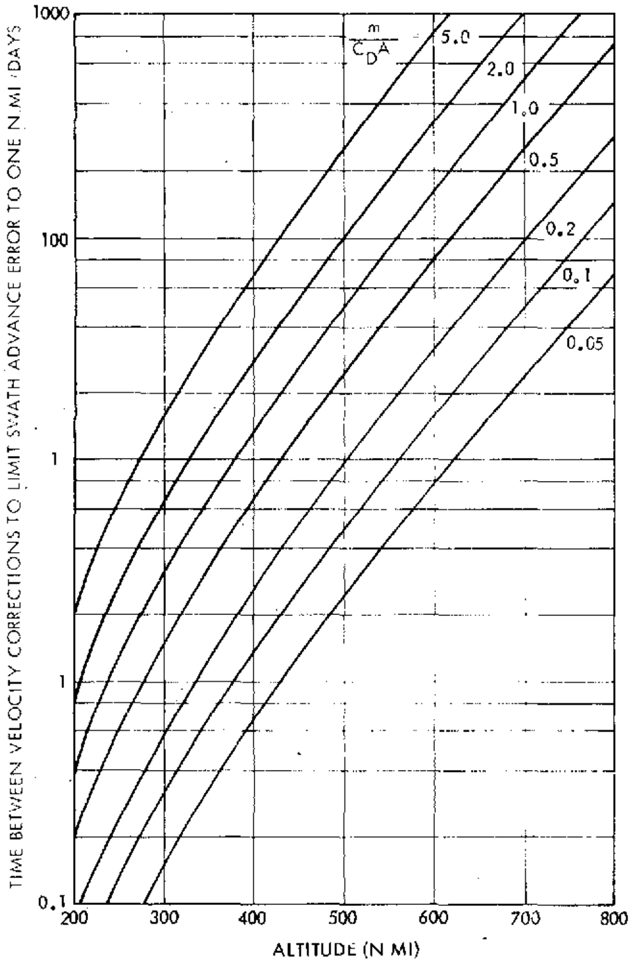


Figure 3-11. Frequency of Velocity Corrections Per Nautical Mile Swath Advance Error

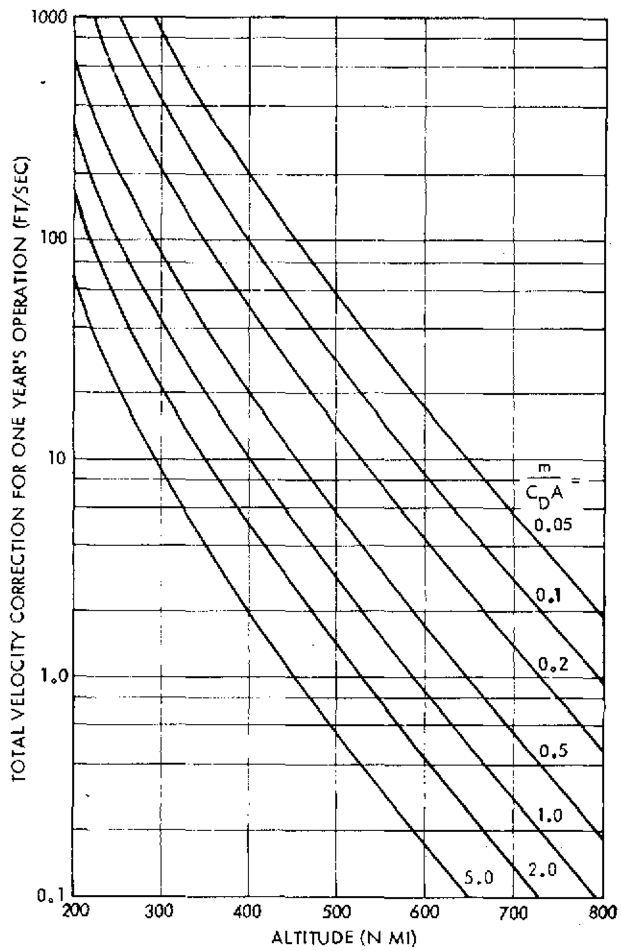


Figure 3-12. Total Velocity Correction Required for 1-Year Operation

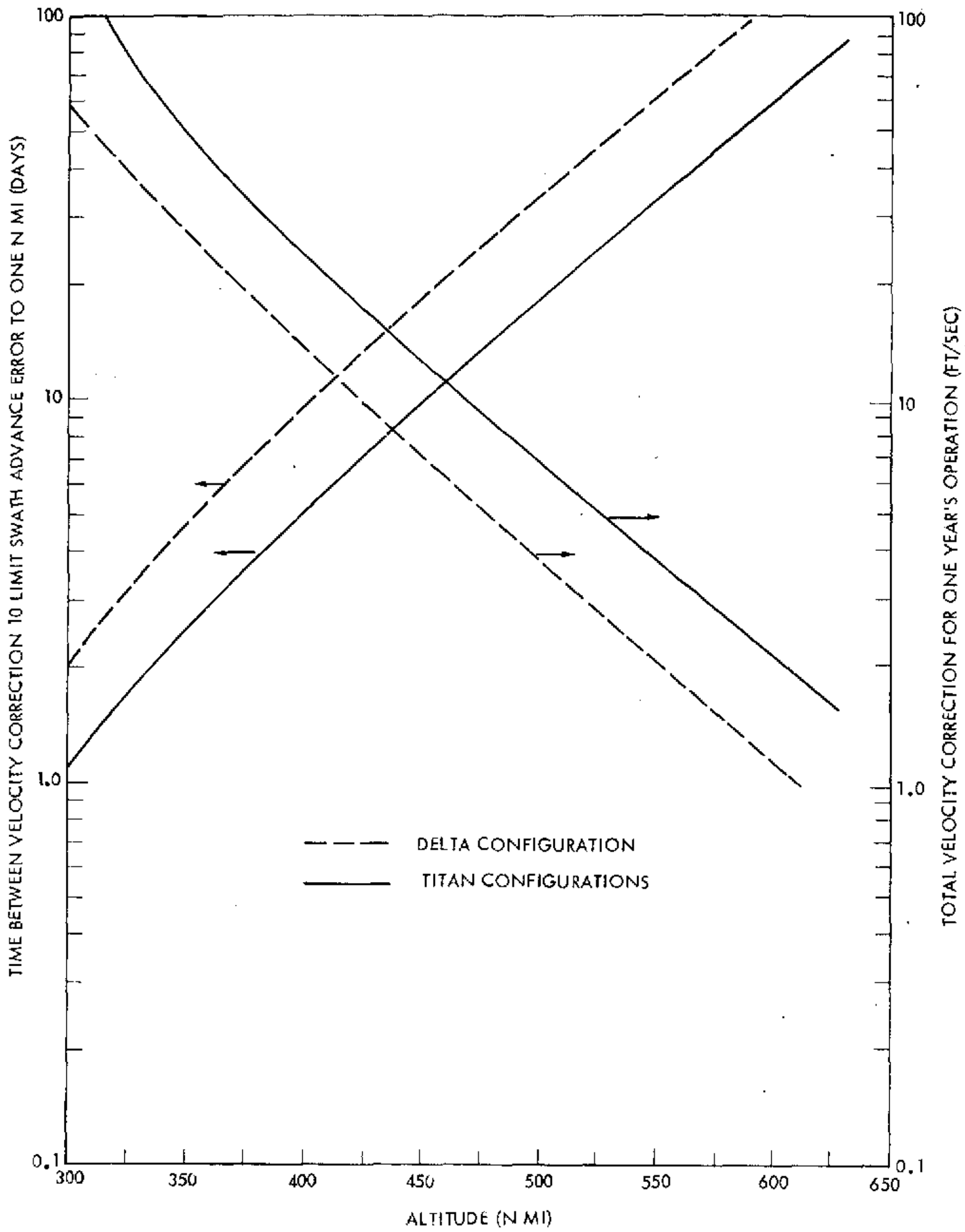


Figure 3-13. Orbit Maintenance for EOS-A

three maneuvers, although two of the actual burns may be combined. The first maneuver is performed to increase apogee to synchronous altitude (ΔV = 8068 ft/sec); the second maneuver will then circularize the orbit (ΔV = 4852 ft/sec). Assuming a due east launch from ETR, a 28.5-degree plane change is required for an equatorial orbit (ΔV = 10,088 ft/sec). The total transfer time from 100 nautical miles is on the order of 5 hours.

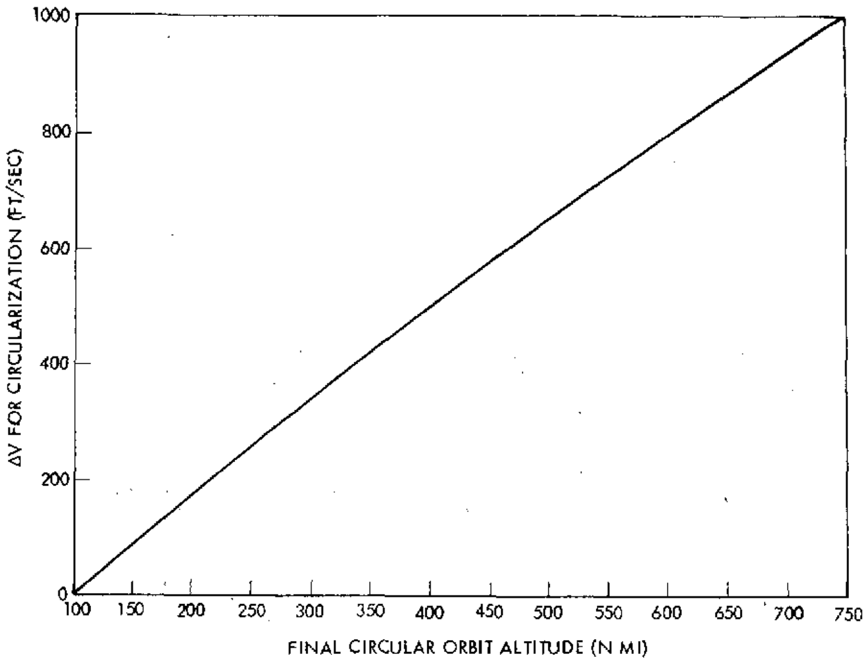


Figure 3-14. Circularization ΔV Requirements

A satellite in an equatorial synchronous orbit experiences acceleration forces due to the inhomogeneity of the earth's gravity field and, to a much larger extent, the influence of the gravity fields of the moon and sun. Consequently, over a long period of time, it will not remain fixed exactly over a given spot on the equator. True maintenance of the satellite on station (e. g., to within 0.1 degree north-south and east-west) will require the use of propellant in a quantity dependent on the satellite position and on the epoch. Typical stationkeeping ΔV requirements are 5 ft/sec in longitude and 150 ft/sec in latitude per year.

3.2 LAUNCH VEHICLE CHARACTERISTICS

Thor-Delta and Titan capabilities for EOS are described in this section. Shuttle performance is discussed in Section 3.3. A subsequent report will treat Shuttle utilization in detail. Additional launch velocity data is presented in Appendix A.

3.2.1 Payload Capabilities

Payload capabilities for the launch vehicle candidates defined in Section 2.3 are shown in Figures 3-15 through 3-17 for several circular orbits and elliptical transfer orbits.

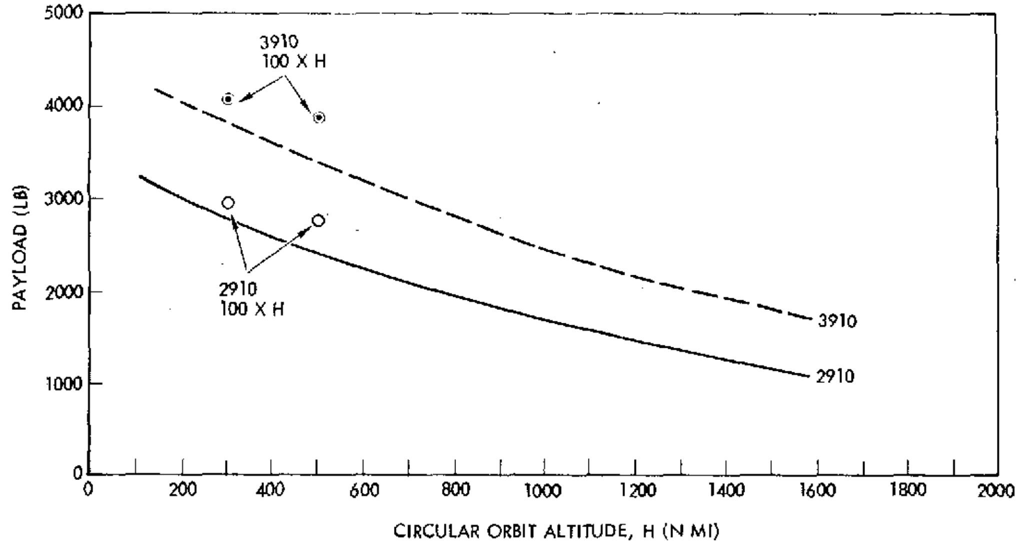


Figure 3-15. Thor-Delta Sun-Synchronous Orbit Capability from WTR

NOTE:
1. 100 N MI INJECT ALTITUDE
2. PERFORMANCE MARGIN IN STAGE II
3. PAYLOAD FAIRING JETTISONED AT 400,000 FT AT WTR

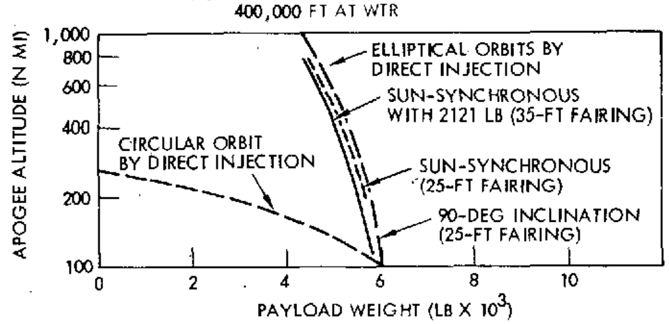


Figure 3-16. Titan IIIB (SSB), Payload Weight Versus Altitude (WTR)

NOTE:
1. 100 N MI INJECT ALTITUDE
2. PERFORMANCE MARGIN IN STAGE II
3. 45-FT PAYLOAD FAIRING JETTISONED AT 400,000 FT

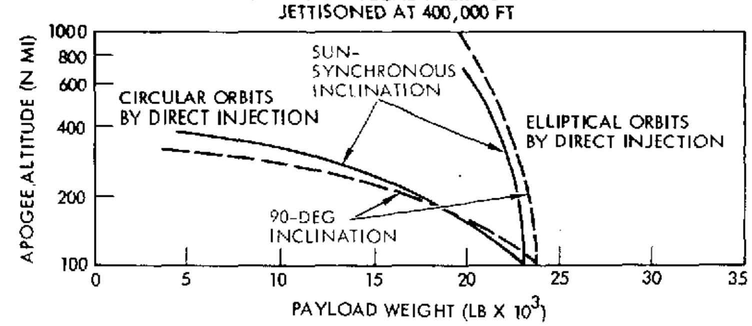


Figure 3-17. Titan IIID Payload Weight Versus Altitude (WTR)

Both the Thor-Delta 2910 and 3910 capabilities for sun-synchronous circular orbits are shown in Figure 3-15. Also spotted on this figure are the capabilities for 100 x 300 and 100 x 500 nautical miles sun-synchronous inclination at apogee altitude. The additional capability to the elliptical orbits is not sufficient to overcome the spacecraft weight penalty associated with the on-board propulsion required to circularize at apogee. Therefore, it is concluded that for Thor-Delta, launching directly into a circular orbit gives the best performance.

The Titan IIIB capabilities for 90 degrees inclined elliptical and circular orbits and elliptical orbit capabilities at sun-synchronous inclinations are shown in Figure 3-16. Note that this launch vehicle is incapable of direct injection to circular orbits above 250 nautical miles. However, elliptical orbit capability to sun-synchrous orbits is significantly greater than those for Thor-Delta, even considering that final circularization propulsion must be subtracted from the net capability.

Figure 3-17 shows the corresponding capabilities for Titan IIID. This vehicle is unable to go directly to circular orbits above about 350 nautical miles, but it has very large capability to elliptical orbits.

For a geostationary orbit, as required by SEOS, the launch vehicle would place the payload into a geostationary transfer orbit (100 x 19,323 nautical miles). Table 3-3 shows the payload capabilities to geostationary transfer orbit for several launch vehicles. To circularize, the additional velocity requirement is 6080 ft/sec, for synchronous equatorial; the propellant required to accomplish this is approximately 50 percent of the payload to transfer orbit, and may be in the form of an additional stage or as part of the Observatory. Other Titan vehicles with greater capabilities are available, at correspondingly higher costs. Launch vehicle adapters must also be subtracted from these capabilities.

The Shuttle capabilities to synchronous equatorial orbits are discussed in Section 3.3.

3.2.2 Payload Fairings and Envelopes

The standard Thor-Delta fairing and its spacecraft envelope are shown in Figure 3-18. The 86-inch diameter restriction applies to the shoulder where the forward ogive begins, but is generally applied down the length of the fairing. At the base, where the fairing is attached to the launch vehicle, the internal envelope is approximately 91.5, accounting for fairing thickness and permissible out-of-roundness.

Table 3-3. Payload to Geostationary Transfer Orbit

	Transfer (lb)
Thor Delta 2910	1500
Thor Delta 3910	2000
Titan IIIB/Burner II	1650
Titan IIIB/Agenda	3400
Titan IIIE*	9600

*The Titan IIIE is similar to the Titan IIID except it is launched from ETR.

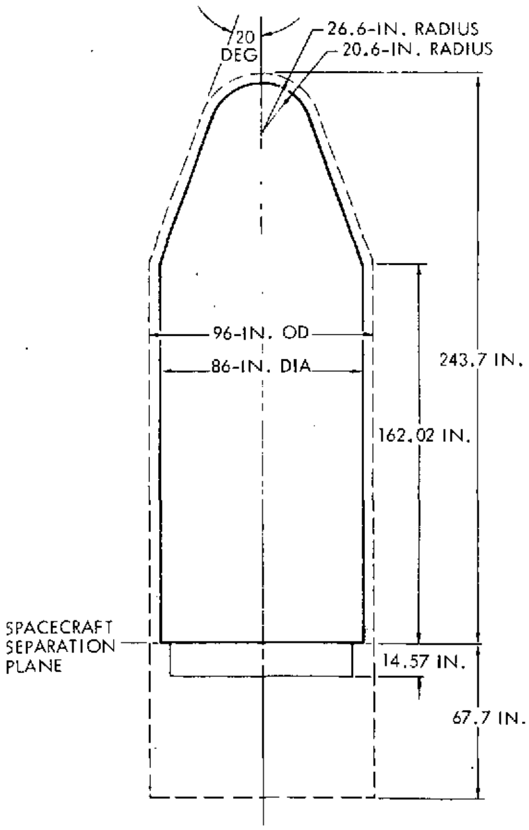


Figure 3-18. Thor-Delta Fairing and Payload Envelope

For the Titan vehicles, either the Lockheed Missile and Space Company P-123 fairing or the McDonnell Douglas Titan IIIC fairing may be used, each with appropriate adapters, for either the Titan IIIB or Titan IIID. These fairings are shown in Figures 3-19 and 3-20. The lengths of these fairings are variable: one or more cylindrical sections may be deleted, starting at the forward end, to derive a fairing of desired length.

3.2.3 Dispersions

Vehicle accuracies at perigee for Titan IIIB and at circular orbit altitude for Thor-Delta are shown in Table 3-4.

3.2.4 Environmental Quasi-Static Limit Loads

Maximum estimated flight loads experienced by primary structure (not including appendages nor equipment panels) are presented in Table 3-5 for Shuttle, Thor-Delta, and Titan launch vehicles. These loading conditions, suitably factored to provide design margin, may be used for preliminary sizing of the primary structure.

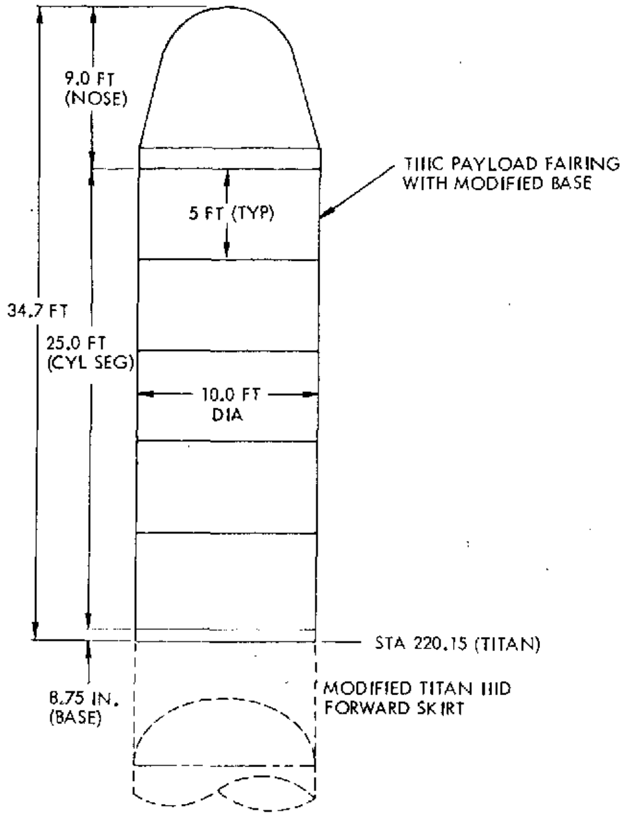


Figure 3-19. Payload Fairing and Burner Shroud for Titan IIID/ Growth Burner (ETR)

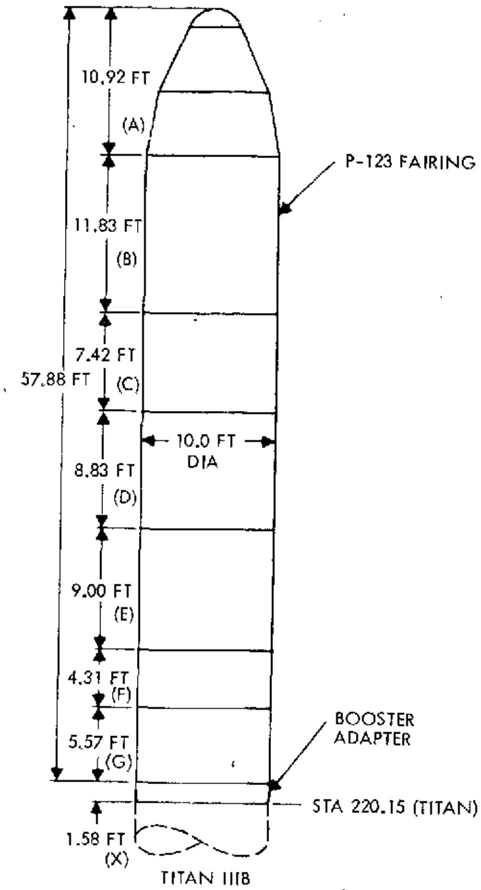


Figure 3-20. Payload Fairing and Agena Shroud for Titan IIIB (SSB)/ Ascent Agena (ETR and WTR)

Table 3-4. Launch Vehicle Dispersion Data (Typical)

Launch Vehicle	Radius (ft)	Inertial Velocity (ft/sec)	Flight Path Angle (ft)	Cross-Range Error (n mi)
T-IIIB	±3700	±20	±0.10	±0.5
2910	±75000	±23	±0.04	--

3.2.4.1 Acoustic Environment

Acoustic environments for various boosters differ in overall level and spectral distribution (Figure 3-21). For Titan, the qualification level is 145 dB overall. For the Thor-Delta 3910, with an acoustically

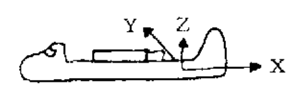
Table 3-5. Maximum Expected Flight Loads (g's)[+*]

Flight Event	Shuttle			Thor-Delta 2910			Titan IIIB 5,000 lb Spacecraft			Titan IIID 15,000 lb Spacecraft		
	X	Y	Z	X	Y	Z	X	Y	Z	X	Y	Z
Liftoff	-2.3	±0.3	-0.8	-2.9 +1.0	±2.0	±2.0	-2.3 +1.0	±2.0	±2.0	-2.5 +1.0	±2.5	±2.0
High or maximum dynamic pressure	-2.0	±0.5	±0.6									
Booster or Stage I burnout	-3.3	±0.2	-0.4	-12.3 -4.0	±0.65	±0.65	-8.2 +2.5	±1.5	±1.5	-7.6 +2.5	±1.5	±1.5
Orbiter or Stage II burnout	-3.3	±0.2	-0.75				-10.8 +2.0	±1.5	±1.5	-6.7 +2.0	±1.0	±1.0
Shuttle space operations	-0.2 0.1	±0.1	±0.1									
Entry and descent	+1.6 -0.25	±1.5	+3.0 -1.0									
Landing and braking	±1.5	±1.5	+2.5									
Crash[**]	+9 -1.5	±1.5	4.5 -2.0									

[+]Each triad of X, Y, Z loads is applied simultaneously.

[*]X, Y, Z refer to Shuttle axes.

[**]Crash loads are ultimate and used only for satellite support fitting design.



insulated shroud, the estimate is 146 dB overall, compared to 144 dB for the Thor-Delta 2910. The Shuttle environment is estimated to be no more severe than Titan's.

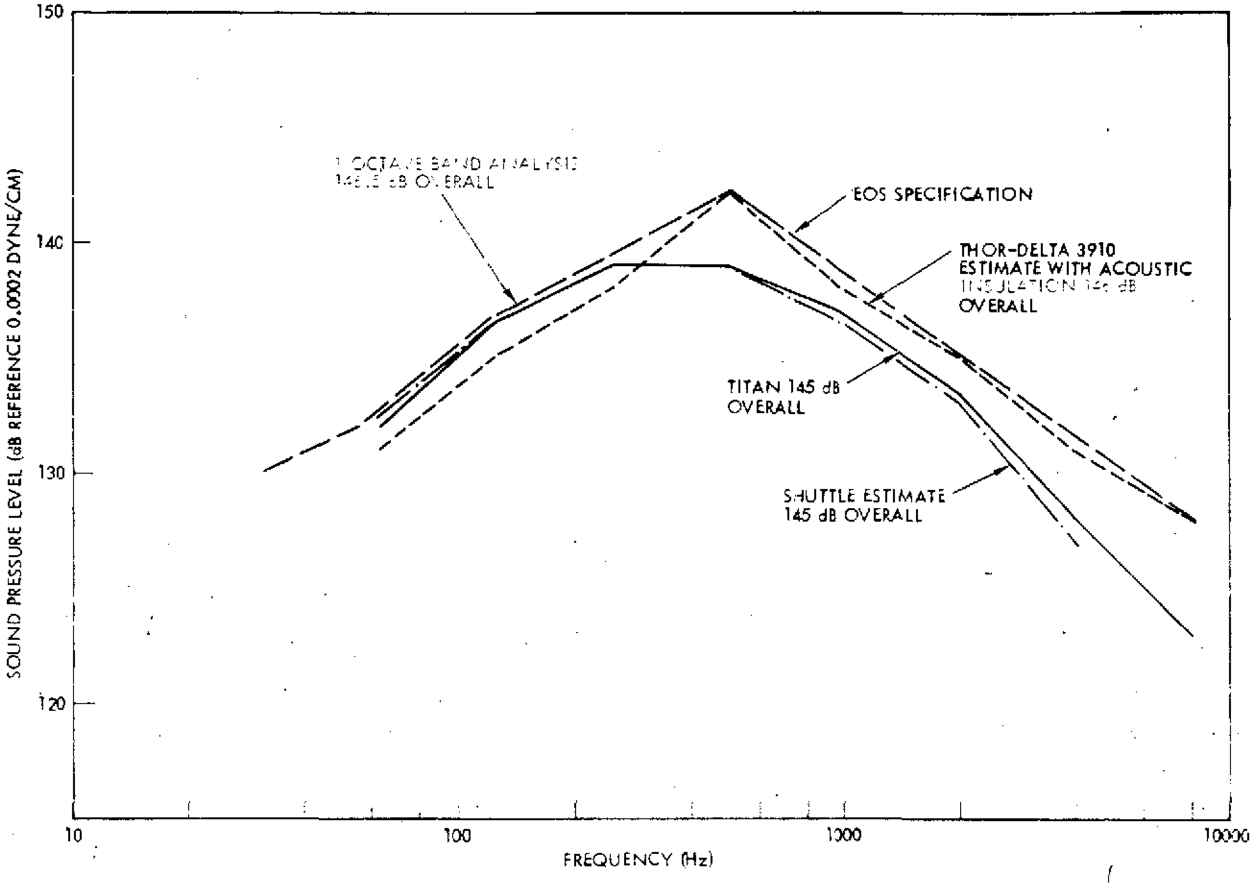


Figure 3-21. EOS Qualification Acoustic Environment

For EOS, an envelope of Thor-Delta and Titan environments is recommended. The spectral distributions of the two are such that this can be done with an overall penalty of only 0.5 dB. This environment is a primary design condition for solar array panels and module radiator surfaces. It also induces the random vibration environment for components within modules.

3.2.4.2 Sinusoidal Environment

The Thor-Delta boosters provide a substantial Pogo environment which is enveloped by the sinusoidal test environments shown in Figure 3-22. Structural response to this environment is used as a design condition, along with the quasi-static loads.

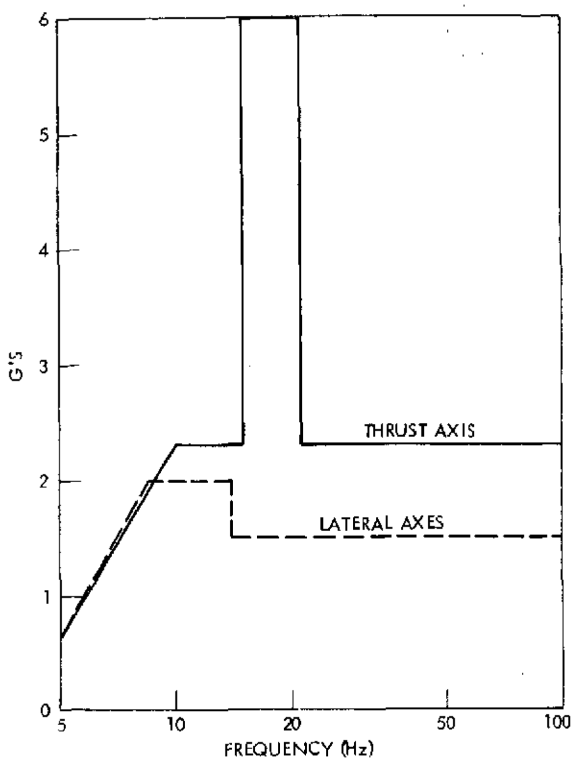


Figure 3-22. Thor-Delta 2910/3910 Sinusoidal Vibration Input

3.2.4.3 Random Vibration Environment

Random vibration environments at the booster/Observatory interface have been defined for Titan, Thor-Delta, and Shuttle. However, these environments are no longer used for design or test on large spacecraft such as EOS, since the interface vibration is heavily dependent on the dynamic characteristics of the spacecraft itself. Acoustic testing is a satisfactory alternative.

3.2.4.4 Shock Environment

The shock environment for EOS Observatories will be dominated by the Observatory separation and deployment ordnance events and will be verified during Observatory qualification. Shock environments induced by booster ordnance actuation are not a design condition.

3.2.5 Costs

The costs of Table 3-6 are based on information received from NASA Headquarters for Thor-Delta, and derived from GSFC and SAMSO data for the Titan vehicles. All are in 1974 dollars.

Table 3-6. Launch Vehicle Costs

	1974 ($M)
Thor-Delta 2910	9.1
Thor-Delta 3910	10.2
Titan IIIB	12.3
Titan IIID	20.5

The cost for the 2910 Thor-Delta is the total mission cost contracted for current launches. For the 3910 Thor-Delta the differential for the larger first-stage solids has been added but only minimum amortization of development is included.

Titan IIIB cost is based on GSFC-provided information, corrected to 1974 dollars. Titan IIID cost is derived by adding the differential between the IIID and IIIB vehicles based on SAMSO data to the IIIB cost.

3.3 SHUTTLE CHARACTERISTICS

Shuttle payload capabilities to circular orbits at various inclinations for both WTR and KSC launches are shown in Figures 3-23 and 3-24. For the highly inclined sun-synchronous orbits, which comprise the majority of the EOS missions, the use of Shuttle for injection to higher circular orbits is inefficient (Figure 3-25). Payload capabilities drop rapidly with altitude; the Shuttle is incapable of attaining an altitude above 490 miles by direct injection at sun-synchronous inclinations. Moreover, to derive net payload capability, the weight of the flight support system (FSS) must be subtracted from the gross payload. The FSS comprises the set of equipment necessary to support the payload within the Shuttle bay, and its current weight is estimated at 5800 pounds by Rockwell International. Thus, direct injection of a 3000 pound spacecraft by the Shuttle to circular sun-synchronous orbits is not feasible for EOS at altitudes above 390 nautical miles (Figure 3-25).*

Like the Titan vehicles, the Shuttle operates much more efficiently into elliptical orbits. Its capability into elliptical orbits whose apogees are at sun-synchronous inclination is shown in Table 3-7, along with the ΔV necessary to circularize at these apogee altitudes. The spacecraft will supply this ΔV by means of an on-board propulsion system.

For EOS altitudes of up to about 1000 nautical miles sun-synchronous, the spacecraft would be equipped with sufficient propulsion capability to circularize at the apogee altitude to which Shuttle would inject. This would require supplemental orbital maneuvering system (OMS) tankage for altitudes above about 500 nautical miles

*Removal of the special-purpose manipulator system (SPMS) for a launch or retrieval only case will reduce the FSS weight by about 3200 pounds, with a proportionate increase in useful payload weight (or altitude).

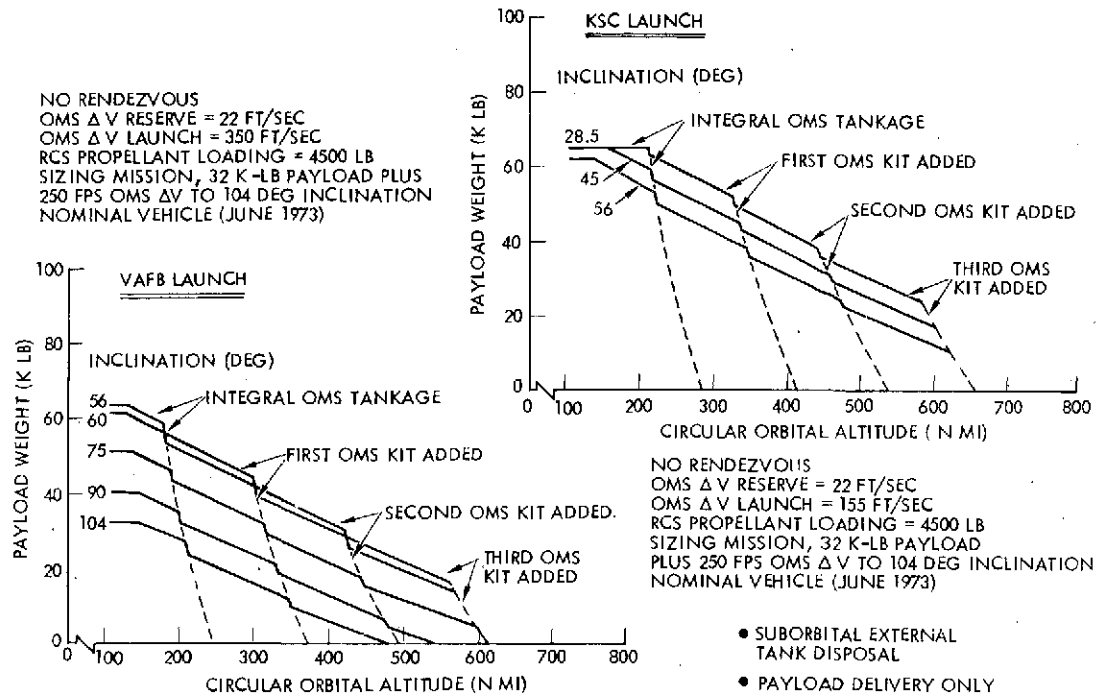


Figure 3-23. Space Shuttle Payload to Circular Orbit

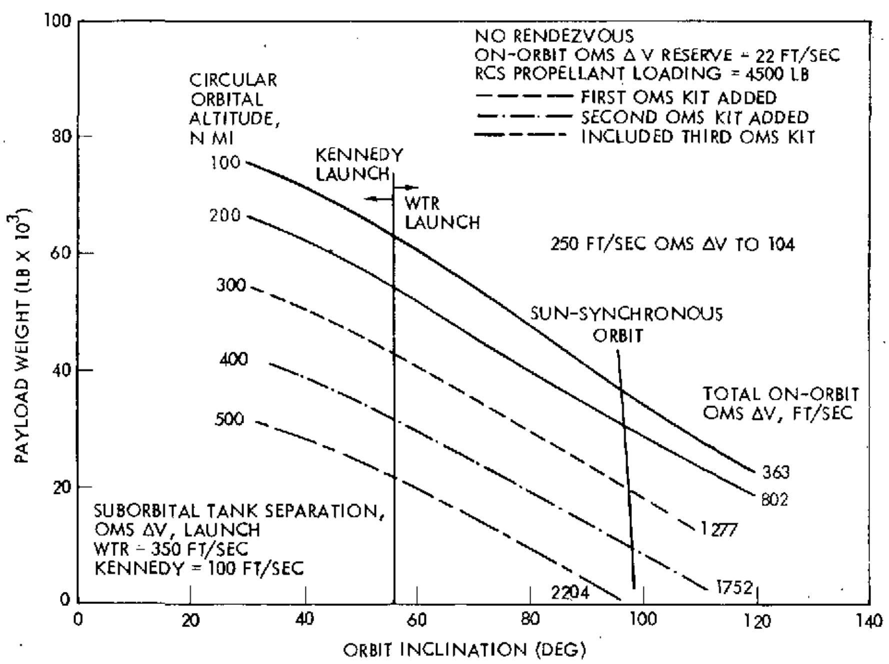


Figure 3-24. Payload Weight Versus Inclination for Various Circular Orbital Altitudes – Delivery Only

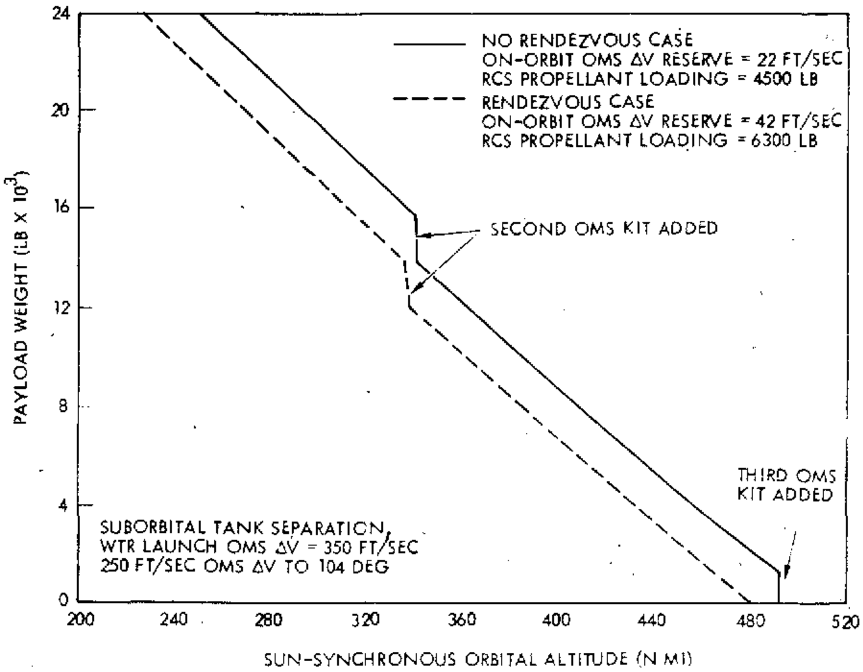


Figure 3-25. Payload Weight Versus Sun-Synchronous Orbital Altitude

Table 3-7. Shuttle Payloads to Sun-Synchronous Elliptical Transfer Orbits (100 Nautical Mile Perigee)*

Altitude (n mi)	Payload Before Circular (lb)	ΔV Circular (ft/sec)
200	42,900	155
300	35,750	340
400	27,700	500
500	19,600	650

For geosynchronous orbits (e.g., SEOS mission) the required ΔV makes spacecraft-integral propulsion not the most cost-effective scheme. Instead a "Tug" is indicated. Table 3-8 gives the payloads to geostationary orbits for several of the Tugs being studied for use with Shuttle. In these cases the integration of the Observatory and Tug must be performed to ensure that the payload bay can accommodate all the necessary equipment. Note that only the Shuttle/Centaur option delivers a large payload into this orbit. Other Tug stages are also being studied.

*All payloads must include flight support system and other payload-peculiar equipment, as well as circularization propellant within the spacecraft.

Table 3-8. Payloads to Geostationary Orbit from ETR with Composite Launch Vehicle

	(lb)
Shuttle/Delta	1, 200
Shuttle/Agena	2, 300
Shuttle/Transtage	2, 800
Shuttle/Centaur	13, 200

For resupply flights with no secondary payload, the Shuttle can service any satellite with which it can rendezvous with the FSS and the replacement modules in its payload bay; the limiting altitude is 410 nautical miles (Figure 3-25). On a dedicated retrieval flight an altitude of 440 nautical miles is serviceable, if the FSS is stripped by removal of the module exchange equipment. For higher sun-synchronous circular orbits the spacecraft will have to deboost into an elliptical rendezvous orbit. This deboost propellant will be included in the spacecraft propulsion complement. Figure 3-26 shows the Shuttle maximum entry payload.

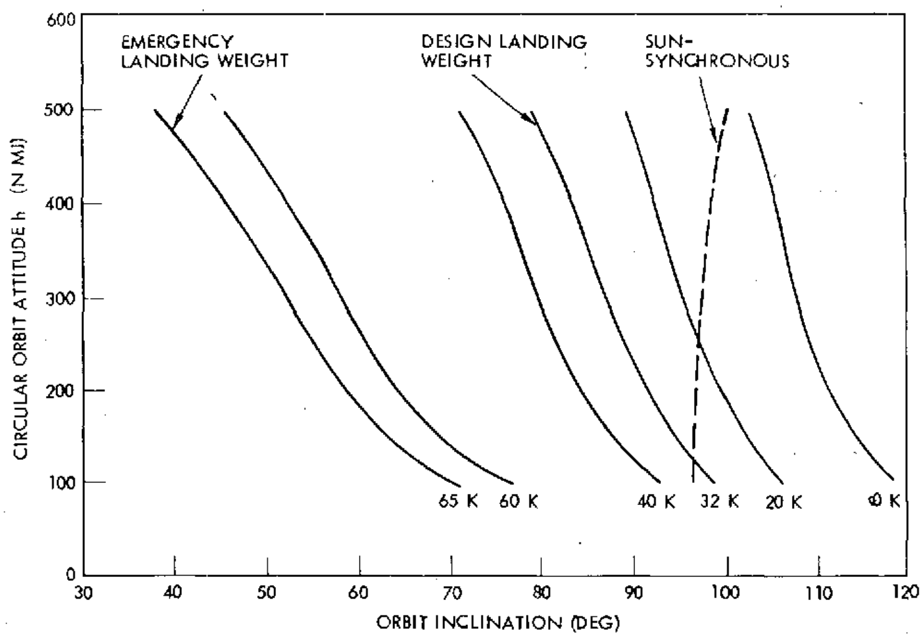


Figure 3-26. Preliminary Direct Reentry Capability

Payload center-of-gravity envelope requirements from Shuttle are shown in Figure 3-27. Shuttle environments are included in Section 3. 2.

For development, as well as retrieval/resupply, EOS will use the Shuttle FSS. The FSS includes accommodation for docking the EOS, a cradle for supporting the EOS at its transition ring, and a SPMS for in-bay module exchange. The EOS is being configured as a modular Observatory, and the SPMS will be compatible with the concept of exchanging modules during servicing Shuttle sorties. The Shuttle-attached equipment, is used for initial contact with the Observatory and to place the Observatory in the docking adapter. The docking adapter is capable of several degrees of freedom; it can place the EOS in the cradle for support during return to earth, and it supports the EOS during module exchange, rotating the Observatory to present the modules being serviced to the SPMS.

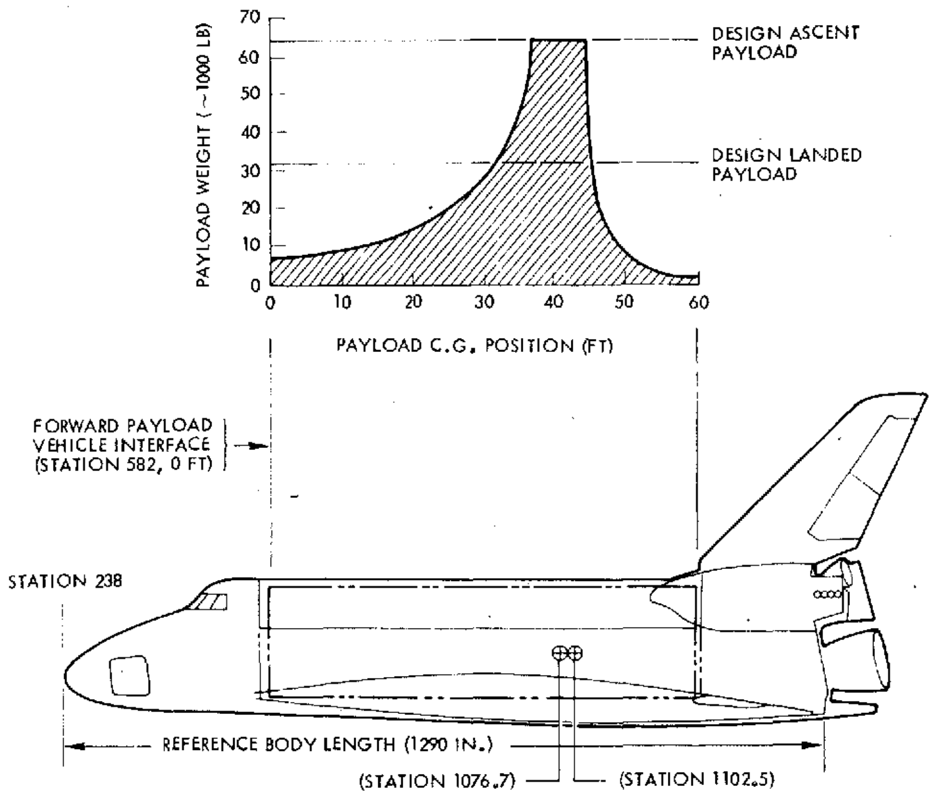


Figure 3-27. Payload Center-of-Gravity Envelope Requirements

4. PAYLOAD DESIGN CONSIDERATIONS

Payload characteristics are significant in orbit and launch vehicle selection. Launch vehicle selection will be constrained by the weight (and size) of the total payload, made up of a selection of the following instruments and other elements. Selection of an orbital altitude can significantly affect the instruments, particularly if a specific performance level (e.g., swathwidth in km) must be achieved independent of orbital altitude.

The following discussion summarizes the dependence of payload elements upon altitude (primarily). Major attention is given to a comparison of the three thematic mapper concepts.

4.1 THEMATIC MAPPER

Weight and performance of the thematic mapper can be significantly influenced by the selected spacecraft altitude. Both performance and cost considerations favor operation at the lower altitudes. Relationships between key instrument parameters and spacecraft altitude are discussed below.

4.1.1 Variation of Weight with Altitude

Groundrules

The following analysis will show, with as few assumptions as possible, the expected variation in weight with altitude of each of the three thematic mapper concepts under study by NASA. The following groundrules are used.

- Swathwidth does not change with altitude
- Ground resolution does not change with altitude
- Although the number of detectors along track may vary as required, the number along scan is limited to one per band.

Since this analysis was performed, several points have been raised which should be incorporated in an updated version of this analysis:

- Honeywell has pointed out that the wide swathwidth calculations assume a flat earth. A curved earth exaggerates the bow-tie effect.

- Honeywell has also supplied a more sophisticated weight equation than that used in the analysis. It is suggested that an equation of this form be used in evaluating all three scanners.

- Te has shown that the 700 microradian limit imposed by image plane aberrations is too severe and have specified some new limits which should be incorporated in the analysis.

- Te also points out that the noise figures used are not consistent. The data used was that supplied by all three manufacturers to NASA in a similar format. It appears that in Te's case this included a cooled detector and an uncooled FET. Further analysis should be done with a cooled FET.

- The Hughes instrument does not make use of beryllium. A substantial reduction in weight might be made by using a beryllium design.

- Further comments and suggestions for additional analysis are expected from Hughes.

Relationship of Weight to Design Parameters

Both Hughes and Te have told us that weight varies approximately as the square of the diameter of the primary mirror. This appears to be a good rule of thumb; however, it should be noted that the weight of the image plane assembly and electronics of the instrument will be independent of the size of the primary mirror. As this represents only 15 percent of the total weight (14 to 16 percent depending on design), it does not have a major effect on the total weight. However, if the number of detectors is increased substantially, it becomes more important. Thus, for purposes of this analysis, 85 percent of the weight is taken to vary as the square of the mirror diameter, and 15 percent to vary as the number of detectors.

Summary of Symbols Used

A	Altitude (km)
a	Aperture area (cm^2)
a_o	Aperture area of reference design

b, c, d, e	Constants
E	Overlap error (pixels)
Δf	Noise bandwidth
I_n	Total RMS noise current
I_o	Noise other than photon noise
I_p	Photon noise current
I_s	Peak signal current
K	Constant
N	Number of detectors per band
N_{max}	Maximum allowable number of detectors per band
N_{opt}	Optimum number of detectors per band
R	Ground resolution
S	Peak signal to RMS noise
T_D	Dwell time
T_S	Scan time
V	Spacecraft ground velocity
W	Swathwidth
w_r	Relative weight
x	Exponent of RMS noise versus bandwidth curve
α	Instantaneous field of view (radians)
η	Scan duty cycle

Effect of altitude on Collecting Aperture and Number of Detectors

The time available for a single scan (T_S) is:

$$T_S = \frac{RN\eta}{V} \tag{1}$$

where:

R is ground resolution

N is number of detectors per band

η is the scan duty cycle

V is the ground velocity.

The dwell time or time available to scan a single pixel (T_D) is:

$$T_D = \frac{R\,T_S}{W} = \frac{R^2 N\eta}{VW} \tag{2}$$

where:

W is the swathwidth

The SNR (S) is represented by the function:

$$S = Ka\alpha^2 T_D{}^x \tag{3}$$

where:

K is a constant

α is the angular field of view

a is the aperture area

x is an exponent to be defined later

and:

$$\alpha = \frac{R}{A} \tag{4}$$

where:

A is the altitude

Combining Equations 1, 2, 3, and 4

$$a = \frac{S}{K} \frac{A^2}{R^{2(x+1)}} \left(\frac{VW}{N\eta}\right)^x \tag{5}$$

For a constant swathwidth, ground resolution, SNR and scan duty cycle:

$$a \approx A^2 \left(\frac{V}{N}\right)^x \tag{6}$$

<u>Method of Determination of the Exponent x</u>

It will be shown that to a good approximation:

$$I_n = C\Delta f^x \tag{7}$$

where:

I_n is the RMS noise

C is the constant

Δf is the noise bandwidth ($\Delta f \approx 1/T_D$)

As $I_n \approx \Delta f^x$, then $S \approx T_D{}^x$ as indicated in Equation 3.

<u>Noise Sources</u>

A good analysis of individual noise terms is given in "Altitude versus Scale Factor, SSR and HRPI Scanners," Report 1064-1, the Te Company, 4 June 1974. This shows that each noise source is a power function of bandwidth, with exponents depending on the individual noise source. These equations have been used with data contained in the comparative data sheets supplied by NASA for all three scanners. The results of the calculations are shown in Table 4-1.

Table 4-1. Variation of Different Noise Sources With Bandwidth*

		Type of Noise						
Noise Bandwidth (kHz)	Thematic Mapper	Photon (S/N = 12)	FET Voltage	FET Current	FET Load Resistance	Feedback Resistance	Detector	RSS Noise
25	Te	14.77	13.37	9.49	2.58	5.25	0.24	22.83
	Honeywell	25.32	9.49	9.49	3.16	6.43	60.40	67.24
	Hughes	36.15	18.48	31.62	3.16	90.99	87.09	136.10
55	Te	35.17	43.63	14.07	8.43	7.79	0.35	58.91
	Honeywell	47.57	30.96	14.07	10.32	9.54	89.59	107.90
	Hughes	66.83	60.32	46.90	10.32	135.0	129.2	212.90
150	Te	115.33	196.47	23.24	37.96	12.87	0.58	232.48
	Honeywell	117.59	139.45	23.24	46.49	15.76	148.0	241.10
	Hughes	160.60	271.68	77.46	46.49	222.9	213.3	450.49

*S/N = 12 with all table entries in amperes x 10^{14}.

Photon Noise

In order to determine the photon noise, it is necessary to find the signal current. The assumption is made that the peak signal to RMS noise ratio is 12.

Then, as:

$$I_p = bI_s^{1/2} \Delta f^{1/2} \tag{8}$$

where:

I_p is photon noise

b is a constant

I_s is signal current

$$I_s = 12\sqrt{I_p^2 + I_o^2} \tag{9}$$

where:

I_o is the RSS of all other noise sources

From Equations 8 and 9:

$$I_s = \frac{(Sb)^2\Delta f + \sqrt{(Sb)^4\Delta f^2 + 4S^2 I_o^2}}{2} \tag{10}$$

$$I_n = \frac{I_s}{S} \tag{11}$$

$$I_p = \sqrt{I_n^2 - I_o^2} \tag{12}$$

Calculation of x

Figure 4-1 is a plot of the RMS noise I_n as a function of bandwidth Δf. The value of the exponent x associated with each design is shown on the curves. It is determined by solving Equation 7 for x using the 55 and

150 kHz points. This gives a residual error at 25 kHz of less than 20 percent in the worst case. It will be used to help solve Equation 6. The different shape of the Te curve is due to the cooled detector giving much lower noise.

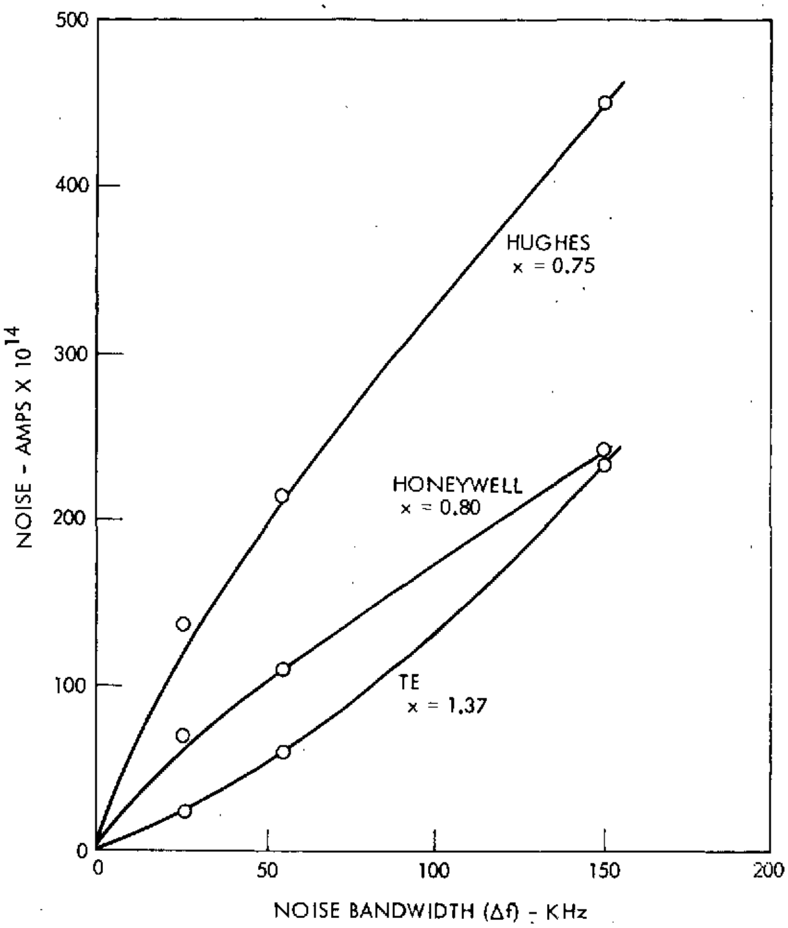


Figure 4-1. Noise Versus Noise Bandwidth

Number of Detectors

In order to minimize a and hence weight, Equation 6 indicates that up to a point N should be a maximum. In the case of the linear scanners (Te and Hughes) the maximum allowable value of N will be determined by the allowable error in scan overlap due to the "bow-tie" effect (spreading of scanline width at off-nadir viewing angles).

It can be shown from geometrical considerations that the maximum number of detectors which may be used with linear scanners is:

$$N_{max} = \frac{E}{\sqrt{1 + \left(\frac{W}{2A}\right)^2} - 1} \qquad (13)$$

where:

E is allowable overlap error in pixels.

In addition to the limit on number of detectors imposed on the Te scanner by the bow-tie effect, an additional limitation is imposed by aberrations in the image plane. The resultant maximum number of detectors is:

$$N_{max} = (7 \times 10^{-4}) \frac{A^*}{R} \qquad (14)$$

The conical scanner does not have this limitation on number of detectors as it operates at a constant path length. In this case the limit is not clearcut. It appears that the first deleterious effect of adding more detectors is a reduction in optical transfer function as the mirror size decreases. Honeywell has indicated that a drop of from 85 to 80 percent in along-track and from 90 to 87 percent in cross-track band 5 MTF at the IFOV frequency will result from a 44 percent reduction in mirror area. This effect will be less for other bands and is so minor that it can be neglected at least until an MTF reduction of about 25 percent occurs. The number of detectors can thus be chosen to minimize the weight at each altitude, with the condition that the mirror area does not drop below about 200 cm^2.

In order to optimize the number of detectors for all scanner types, the weight equation must be derived, differentiated, and equated to zero.

From Equation 6:

$$N = V \left(\frac{eA^2}{a}\right)^{\frac{1}{x}} \qquad (15)$$

---

* See comment on page 4-2.

$$c = \begin{Bmatrix} 0.002415 \text{ for Honeywell} \\ 0.003790 \text{ for Hughes} \\ 0.004184 \text{ for Te} \end{Bmatrix} \text{with V in } \frac{\text{km}}{\text{sec.}}, \text{ A in km, a in cm}^2$$

The relative weight (w_r) is:

$$w_r = \left(\frac{a}{a_o}\right) 0.85 + \left(\frac{N}{16}\right) 0.15 \tag{16}$$

where:

a_o is the baseline design aperture area.

Differentiating, equating to zero, and solving for N gives:

$$N_{opt} = \left(\frac{A^2 V^x}{d}\right)^{\frac{1}{1+x}} \tag{17}$$

$$d = \begin{cases} 5422 \text{ for Honeywell} \\ 3841 \text{ for Hughes} \\ 1202 \text{ for Te} \end{cases}$$

N_{opt} is the number of detectors giving minimum weight. The aperture area (a) can now be obtained from Equation 15.

Variation of Weight Factor with Altitude

Table 4-2 lists the variation of the maximum number of detectors, the optimum number of detectors, and the relative weight (w_r) with altitude for each design. In the case of the Hughes design, the number of detectors at all altitudes up to and including 1300 km is determined by the bow-tie effect (with allowable overlap of 0.15 pixels) as indicated by N_{max}. The Te design is limited by bow-tie at 500 km and Equation 14 at all other altitudes.

Table 4-2 weight tables give variations in weight with altitude, but cannot be used to compare one scanner with another.

Table 4-2. Number of Detectors and Weight Factor Versus Altitude

Altitude (km)	V (km/sec)	N_{max}	N_{max} (Te only)	N_{opt}			Relative Weight (900 km = 1)		
				Honeywell	Hughes	Te	Honeywell	Hughes	Te
500	7.062	8.84	8.84	20.03	25.13	34.18	0.540	0.719	1.008*
700	6.765	17.26	16.33	28.56	36.23	44.28	0.770	0.864	0.882
900	6.488	28.47	21.00	37.06	47.43	53.45	1.000	1.000	1.000
1100	6.229	42.51	25.67	45.48	58.63	61.83	1.227	1.136	1.099
1300	5.988	59.36	30.33	53.81	69.77	69.60	1.452	1.275	1.184
1500	5.761	78.95	35.00	62.01	80.81	76.79	1.673	1.421	1.258
1700	5.548	101.42	39.67	70.08	91.74	83.51	1.891	1.605	1.324
1900	5.349	126.69	44.33	78.02	102.56	89.81	2.105	1.794	1.385
2100	5.160	154.64	49.00	85.81	113.23	95.71	2.284	1.980	1.437

*Limited by bow-tie effect.

Variation of Weight with Altitude

Using the weight factors of the preceding section, and the following manufacturers data, the actual variations of weight with altitude can be obtained.

Manufacturer	Altitude (km)	Weight (lb)
Honeywell	900	350
Hughes	717	320
Te Company	715	325

Table 4-3 gives the comparative weights of all three designs.

Table 4-3. Weight Versus Altitude

Altitude (km)	Weight (lb)		
	Honeywell	Hughes	Te
500	189	263	367
700	270	316	321
900	350	365	364
1100	429	415	400
1300	508	466	431
1500	586	520	458
1700	662	587	482
1900	737	656	504
2100	799	724	523

Figure 4-2 is a plot of these weights.

Table 4-4 gives the weights of the Hughes and Te scanners if no restriction is placed on the allowable bow-tie scan-to-scan overlap. Values not included in the table are unchanged from Table 4-3.

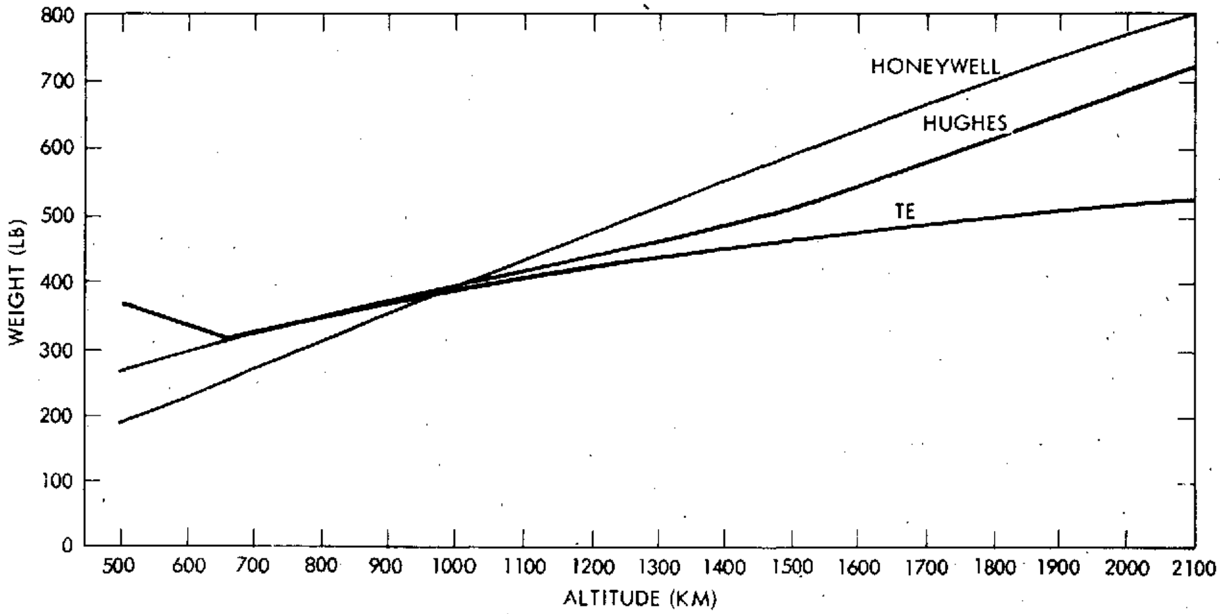


Figure 4-2. Weight Versus Altitude for all Three Concepts

Table 4-4. Weights if Scan Overlap is Ignored

Altitude (km)	Weight (lb)	
	Hughes	Te
500	161	251
700	232	Unchanged
900	304	Unchanged
1100	375	Unchanged
1300	446	Unchanged
1500	517	Unchanged

4.1.2 Extended Swathwidth Capabilities

Groundrules

The assumptions used in the following section are:

- Ground resolution is kept constant at 30 m.
- The Te scanner uses the single-mirror configuration.

Future analysis should cover the following additional cases:

- Te scanner with dual-mirror configuration
- Relaxed ground resolution requirements.

The purpose of this section is to define and evaluate the factors which limit the use of each of the three thematic mapper concepts at wider swathwidths than the nominal 185 km. The study has been done as a function of altitude. Each of the three types will be considered separately.

<u>Hughes Object Plane Linear Scanner</u>

This instrument always operates close to its optical axis and thus has no aberration limit imposed on its field of view in the image plane. The limit on its angular field of view (and hence on swathwidth for a given altitude) is due to the overlap of contiguous scans at off-nadir points ("bow-tie" effect), which is common to all linear scanners either object or image plane.

In the case where swathwidth is allowed to vary, but ground resolution, SNR, and scan duty cycle are held constant, Equation 5 shows that:

$$a \approx A^2\left(\frac{VW}{N}\right)^x \qquad (18)$$

where:

a is aperture area

A is altitude

V is spacecraft velocity

N is number of detectors per band

W is swathwidth

The maximum allowable number of detectors (N_{max}) can be obtained from Equation 13 and is given in Table 4-5 for the case of 0.15 pixel scan-to-scan overlap.

Table 4-5. N_{max} Versus Altitude and Swathwidth

Altitude (km)	Swathwidth (km)			
	185	300	400	500
500	8.84	3.41	1.95	1.27
700	17.26	6.61	3.75	2.42
900	28.47	10.87	6.15	3.96
1100	42.51	16.21	9.15	5.88
1300	59.36	22.61	12.75	8.19
1500	78.95	30.07	16.95	10.87
1700	101.42	38.61	21.75	13.95
1900	126.69	48.21	27.15	17.40
2100	154.64	58.87	33.15	21.24

As shown, the optimum number of detectors can be determined by combining Equation 18 with Equation 16, differentiating, equating to zero, and solving for N_{opt}.

The value of N_{opt} is given by:

$$N_{opt} = \left(\frac{A^2 V^x W^x}{e}\right)^{\frac{1}{1+x}} \qquad (19)$$

$$e = \begin{cases} 3.530 \times 10^5 & \text{for Honeywell} \\ 1.926 \times 10^5 & \text{for Hughes} \\ 15.33 \times 10^5 & \text{for Te} \end{cases}$$

Tabulated values of N_{opt} for the Hughes scanner are given in Table 4-6. When compared with the values of N_{max} in Table 4-5, it is apparent that $N_{opt} > N_{max}$ at all swathwidths considered except 185 km. Thus N_{max} must be used as the number of detectors, new apertures calculated, and the weight calculated from Equation 16.

Table 4-6. N_{opt} for Hughes Scanner Versus Altitude and Swathwidth

Altitude (km)	Swathwidth (km)			
	185	300	400	500
500	25.13	30.92	34.97	38.48
700	36.23	44.57	50.42	55.48
900	57.43	58.35	66.00	72.63
1100	58.63	72.13	81.59	89.78
1300	69.77	85.83	97.09	106.84
1500	80.81	99.41	112.46	123.74
1700	91.74	112.86	127.67	140.48
1900	102.56	126.17	142.73	157.05
2100	113.23	139.30	157.57	173.39

Table 4-7 gives the Hughes scanner weights as a function of swathwidth and Figure 4-3 shows the same data graphically.

Table 4-7. Hughes Scanner Weight Versus Swathwidth

Altitude (km)	Weight (lb)			
	Swathwidth (km)			
	185	300	400	500
500	263	343	419	490
700	316	415	502	584
900	365	485	581	672
1100	415	556	660	756
1300	466	631	740	842

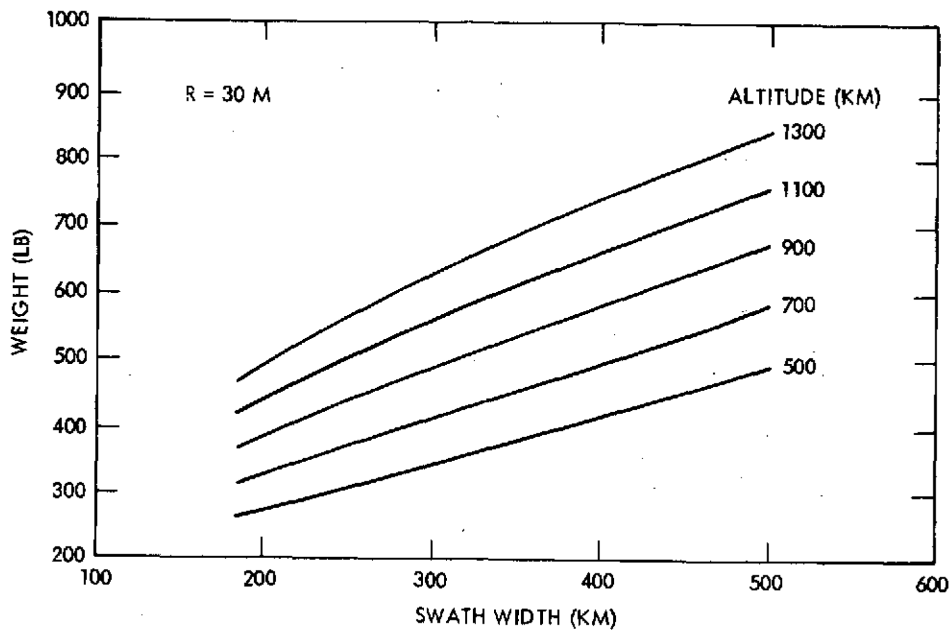


Figure 4-3. Weight of Hughes Scanner

Honeywell Image Plane Conical Scanner

This instrument is limited in swathwidth by image plane aberrations. The equation governing the maximum swathwidth is:

$$W_{max} = 2A \sin \frac{\beta}{2} \tan \gamma_{max} \tag{20}$$

where:

$$\beta = \frac{2\pi\eta}{M}$$

M = number of mirrors

η = scan efficiency

γ_{max} = maximum half cone angle for given resolution

Honeywell gives values of γ of 17 degrees and 25.3 degrees for resolutions of 33 and 56 microradians, respectively. A linear interpolation then gives:

$$\gamma_{max} = 6519 \frac{R}{A} + 0.0794 \tag{21}$$

with all angles in radians

$$\rho = \frac{R}{A} \tag{22}$$

where:

R = ground resolution

Combining the preceding three equations gives:

$$W_{max} = 2A \sin\left(\frac{\pi\eta}{M}\right) \tan\left[6.519 \frac{R}{A} + 0.0794\right] \tag{23}$$

with A in km.

Solving this equation for 30-meter ground resolution gives the data of Table 4-8.

N_{opt} for the Honeywell scanner can be solved from Equation 2 and is listed in Table 4-9.

As the Honeywell scanner is not limited by the bow-tie effect, N_{opt} may be used in each case. The aperture areas required to maintain the SNR are calculated by the method described earlier and then the weights are determined by Equation 16. (See Table 4-10.)

Table 4-8. Honeywell Scanner Maximum Swathwidth Versus Altitude

Altitude (km)	Swathwidth (km)			
	Number of Mirrors			
	3	4	5	6
500	378	299	245	207
700	390	309	253	214
900	409	323	265	224
1100	430	340	279	235
1300	452	358	293	247

Table 4-9. N_{opt} for Honeywell Scanner Versus Altitude and Swathwidth

Altitude (km)	Swathwidth (km)			
	185	300	400	500
500	20.03	24.83	28.22	31.16
700	28.56	35.41	40.23	44.43
900	37.06	45.94	52.21	57.65
1100	45.48	56.38	64.07	70.75
1300	53.81	66.71	75.81	83.71

Table 4-10. Honeywell Scanner Weight Versus Swathwidth

Altitude (km)	Weight (lb)			
	Swathwidth (km)			
	185	300	400	500
500	189	234	266	294
700	270	334	380	419
900	350	434	493	544
1100	429	533	605	668
1300	508	630	716	791

These weights are shown in Figure 4-4. The maximum swathwidth due to the aberration limit is also shown on this figure. This limit will improve as the resolution is allowed to degrade from 30 meters.

Te Image Plane Linear Scanner

The limit on the angular field of this scanner can be either due to bow-tie effect or to image plane aberrations. The aberration limit is about 700 microradians in the vehicle track direction. In this case:

$$N_{max} = 700 \times 10^{-6} \times \frac{A}{R} \tag{24}$$

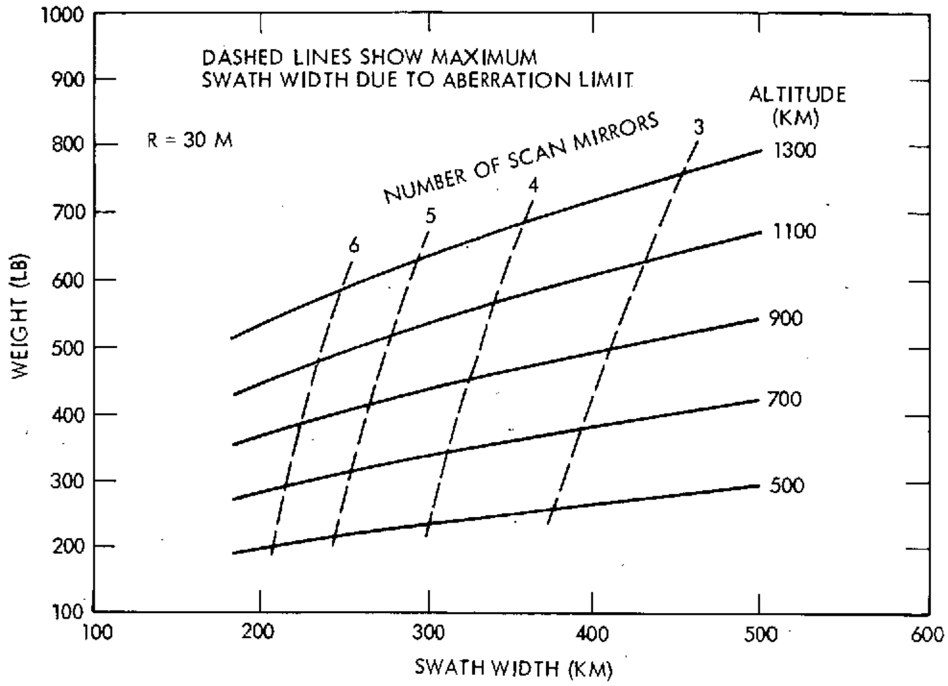


Figure 4-4. Weight of Honeywell Scanner

In the cross-track direction, another limit is imposed by roof mirror vignetting. This limit controls the primary aperture size, as increased aperture will cause roof mirror vignetting and/or a lower scan efficiency.

To a first approximation, the maximum number of roofs can be determined from:

$$M = M_o \frac{\delta_o}{\delta} \tag{25}$$

where:

M is the number of roofs

M_o is the number of roofs in the reference design

δ is the angular field of view

δ_o is the angular field of view of the reference design

It can also be shown that:

$$\eta = \frac{M\delta}{2\pi} \tag{26}$$

and hence:

$$\eta = \frac{M_o \delta_o}{2\pi} \tag{27}$$

indicating that as the number of roofs is decreased in order to prevent vignetting at wider swaths, the scan efficiency will not change.

However, as mentioned above, an increase in aperture can force a reduction in the number of mirrors without a corresponding increase in angular coverage, and this will result in a lower scan efficiency (see Equation 26).

Assuming that the width of the converging bundle of rays at the edge of the scan pattern is equal to one half of the width of a half roof, then it can be shown that:

$$\frac{M}{M_o} = \frac{4}{3 + \sqrt{\frac{a}{a_o}}} \tag{28}$$

As in previous equations for number of detectors, this equation allows the number of mirrors to take non-integer values for purposes of analysis.

Because:

$$\delta = 2 \tan^{-1}\left(\frac{W}{2A}\right) \tag{29}$$

combining Equations 26, 28, and 29 gives:

$$\eta = \frac{4M_o \tan^{-1}\left(\frac{W}{2A}\right)}{\pi\left(3 + \sqrt{\frac{a}{a_o}}\right)} \tag{30}$$

Equation 5, when combined with Equation 30 for the case of constant ground resolution and SNR, gives:

$$N = \left(\frac{JA^2}{a}\right)^{\frac{1}{x}} \frac{\left(3 + \sqrt{\frac{a}{a_o}}\right) VW}{\tan^{-1}\left(\frac{W}{2A}\right)} \tag{31}$$

where:

$$J = 7.254 \times 10^{-6} \text{ with A and W in km, a in } cm^2, \text{ V in } \frac{km}{sec}$$

with this scanner, the number of detectors per channel is limited by bow-tie effect at swathwidths of 300 km and greater, and by Equation 24 at a swathwidth of 185 km (at all altitudes over 660 km). The area corresponding to the given number of detectors is obtained from Equation 31. The weight is then calculated from Equation 16 and is given in Table 4-11.

The very large increase in weight even at a swathwidth of only 300 km indicates that the single-scan Te scanner would not be the choice for requirements with increased swathwidth.

Table 4-11. Weight of Te Scanner

Altitude (km)	Swathwidth (km)	
	185	300
500	367	1001
700	321	1280
900	364	1436
1100	400	—
1300	431	—

4.1.3 Cost Versus Altitude

NASA personnel have evolved an econometric model, based on eight multispectral scanner type instruments, which provides an estimate of the dependence of cost on altitude-dependent parameters:

$$C = 0.423 \cdot (EU)^{0.6} \cdot (Chan)^{0.57} \cdot (IFOV)^{-0.13}$$

where

C = program cost in millions of dollars

EU = number of equivalent units:

- 1.0 for one flight unit
- 4 to 7 for development
- 1.5 for prototype unit
- 0.5 for prototype refurbishment

Channel = number of information channels = number of detectors

IFOV = instantaneous field of view (millirad)

Using the design parameters from the minimum distortion configuration, the cost versus altitude is indicated in Table 4-12. These data indicate that the total instrument program cost at the 1100-km altitude is almost double the 600 km figure. Relative cost estimates have been obtained from both the Te Company and Hughes as a function of altitude (Figure 4-5). Differences in altitude-cost dependence relates to the differing design criteria in these cases (e.g., number of detectors). Data obtained from the Te Company indicates only a 2 percent increase in instrument cost for the total 600- to 1100-km altitude range. All estimates agree that the lower altitude area is best from a cost standpoint. At most a 20-percent cost penalty is paid for operation at the baseline 717 versus 600 km.

Table 4-12. TM Cost Versus Altitude

Parameters	Altitude (km) 600	717	914	1100
Equivalent units	7 to 10	7 to 10	7 to 10	7 to 10
Information channels	69	100	168	244
IFOV (milliradian)	0.0500	0.0418	0.0328	0.0273
Cost ($M)	22.4 to 27.8	28.4 to 35.2	39.3 to 48.8	49.8 to 61.8

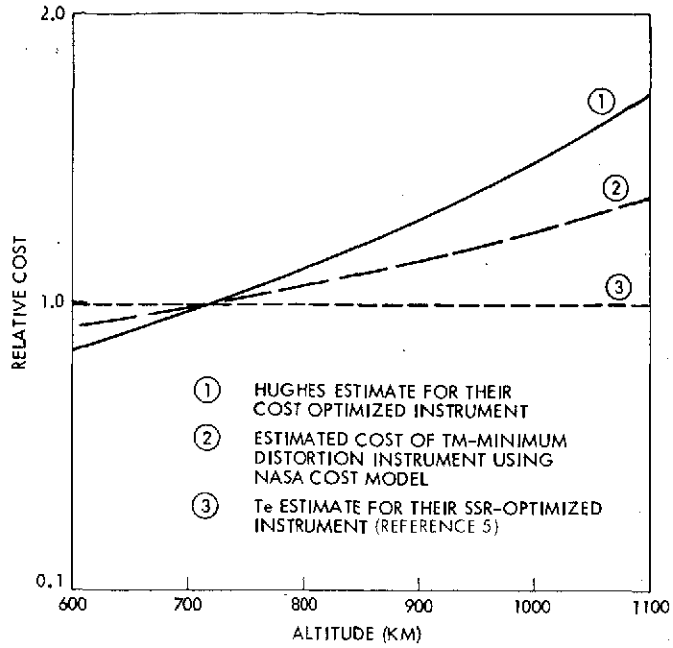


Figure 4-5. Manufacturers Cost Estimates

4.2 HIGH RESOLUTION POINTABLE IMAGER

Relationships between instrument design, cost, and altitude are given below for the selected baseline HRPI that was proposed by the Westinghouse Electric Corporation. Typical conclusions are that an altitude decrease from 917 to 714 km results in a total instrument weight reduction of 106 pounds (436 reduced to 330 pounds) and a 10 percent savings in instrument cost ($1.6 M). Except for minor viewing and data rate effects, altitude change has little effect upon HRPI performance (assuming constant f_{no} optics and constant ground resolution). Weight and cost estimates as a function of altitude are also given for the alternate HRPI configurations (mechanical scanners proposed by Honeywell, Hughes, and Te).

4.2.1 Design Parameters Versus Altitude

4.2.1.1 Viewing Parameters

The following viewing parameters were calculated as a function of spacecraft altitude, h:

5. Te Company Report Number 10644-1, "SSR and HRPI Cost Impact Study," June 14, 1974.

- Offset pointing range, d (Figure 4-6)
- Viewing obliquity, α (Figure 4-6)
- Offset pointing ground distance from nadir, x (Figure 4-6)
- Offset pointing swathwidth, s (Figure 4-7)
- Resolution element footprint dimensions, $\ell \times w$ (Figure 4-8).

Offset distance capability increases from 350 at 600 km altitude to 650 at 1100 km. This can have significant impact on coverage frequency (Section 3.1). Swathwidth, obliquity, and resolution footprint dimensions are all weak functions of altitude (Table 4-13).

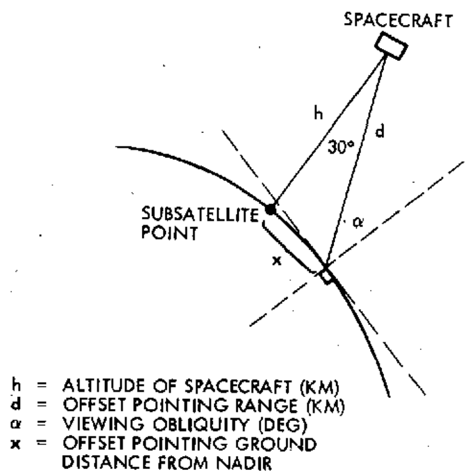


Figure 4-6. Viewing Parameters (See Table 4-13 for d, α, and x as a Function of h)

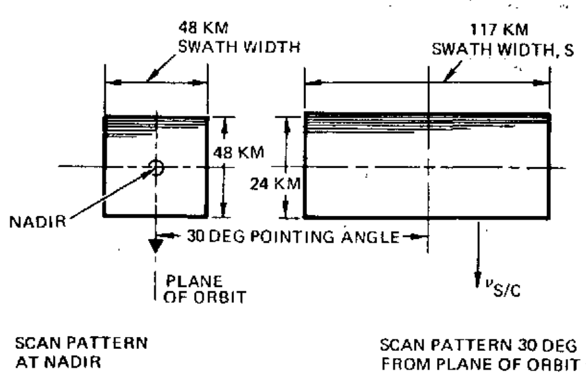


Figure 4-7. HRPI Scan Pattern at Nadir and 30-Degree Offset Pointing (See Table 4-13 for s as a Function of Altitude)

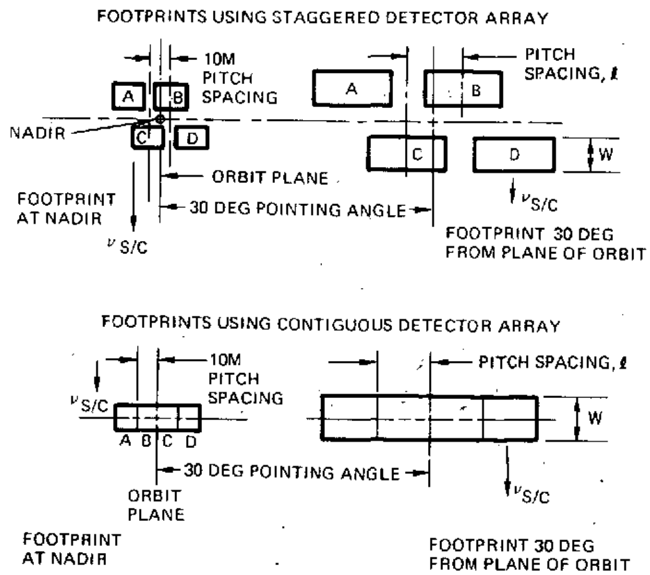


Figure 4-8. HRPI Resolution Element Footprint at Nadir and at 30-Degree Offset Pointing (See Table 4-13 for w and ℓ as a Function of Altitude) — Westinghouse Sensor Configuration

Table 4-13. Viewing Parameters as a Function of Spacecraft Altitude for 30 Degree HRPI Offset Pointing

Altitude h (km)	Offset Pointing Range d (km)	Offset Distance from Nadir x (km)	Offset Pointing Swath Width s (km)	Viewing Obliquity α (deg)	Resolution Element Dimensions	
					ℓ (m)	w (m)
600	704	352	68.2	33.2	14.0	11.7
717	844	422	68.9	33.8	14.2	11.8
914	1082	541	70.2	34.9	14.4	11.8
1100	1309	655	71.4	35.9	14.7	11.9

Data Rate

The HRPI ground resolution and swathwidth remain constant with altitude. Therefore, the sensor data rate variation with altitude is simply an inverse function of line-frame time, t_f. The line-frame time is given by:

$$t_f = \frac{w}{v} \text{ (seconds)} \quad (33)$$

where

w = ground resolution (10 meters)

v = suborbital ground track velocity (meters/sec).

The sensor data rate is given by

$$\text{data rate} = \frac{n_\ell \cdot b \cdot \eta_b}{t_f \cdot \phi_s} \text{ (bits/sec)} \quad (34)$$

where

n = number of resolution elements per scan line (4864)

b = number of bits per data sample work (8)

η_b = number of spectral bands (4)

ϕ_s = detector sampling duty cycle (0.82).

The parameters v and t_f, as well as sensor data rate, are given as a function of spacecraft altitude in Table 4-14. Data rate decreases with altitude at a rate of 2 percent per 100 km over the altitude range 600 to 1100 km.

Table 4-14. HRPI Data Rate Versus Altitude

Altitude (km)	V_{sat} (m/sec)	t_f (msec)	Data Rate (Mbit/sec)
600	6909	1.447	131
717	6739	1.484	128
914	6468	1.546	123
1100	6228	1.606	118

Size and Weight

The size and weight analysis assumed the following:

- Detector size and image plane spacing remains fixed with altitude*
- The telescope focal length is proportional to altitude, thus maintaining a constant 10-meter ground resolution and 48-km swathwidth
- The telescope aperture diameter varies with altitude to maintain a constant f/3.0 optical system (i.e., constant image plane irradiance).

This results in maximum size and weight reduction with reduced altitude and nearly constant sensor SNR performance. The driving forces for weight change with altitude are telescope and pointing mirror size requirements. The weight estimates assume the use of beryllium optical and structure components where feasible.

Table 4-15 summarizes the size and weight of the HRPI components as a function of design altitude. Variations in cross-track sensor length account for changes in telescope and pointing mirror length. Nadir-axis sensor height must increase with altitude to allow for full swing of the

*In order to minimize optical dimensions (and therefore sensor weight), detector size is minimized to state-of-the-art capability regardless of altitude.

pointing mirror. Only small changes in along-track sensor width are necessary since that dimension depends on electronics packaging requirements rather than optical dimensions.

Table 4-15. HRPI Component Size and Weight Versus Altitude

Parameter	Design Altitude (km) 600	717	914	1100
Size parameters (in.)				
Optical focal length	36	43	54.8	66
Telescope physical length	13	17	22	26
Telescope aperture diameter	12	15	18.3	22
Pointing mirror dimensions (oval)	13 x 26	15 x 30	19 x 38	23 x 46
Cross-track sensor length	66	72	84	94
Nadir-axis sensor height	27	30	37	43
Along-track sensor width	24	25	26	27
Weight parameters* (lb)				
Telescope assembly (h)	44	70	132	213
Detectors and beam splitter	8	8	8	8
Mirror and pointing assembly	48	54	68	95
Angular momentum compensation	19	19	19	19
Electronics	28	28	28	28
Thermal control	16	16	16	16
Miscellaneous	20	20	20	20
Structure	80	85	105	130
Net	263	300	396	526
Contingency (10%)	26	30	40	53
Total	289	330	436	579

*Beryllium structure and pointing mirror.

4.2.2 Weight/Cost Summary

Figure 4-9 illustrates the change in HRPI weight with altitude (Westinghouse configuration) along with the weight savings afforded by using beryllium components instead of conventional materials. The electronically scanned Westinghouse HRPI is the selected baseline design for the EOS-A satellite (see Reference 2, Section 5.2) and has been given detailed attention in the weight versus altitude study. For completeness, summary weight versus altitude curves for the alternate mechanical scanner designs are given in Figure 4-10. In comparing the electronic

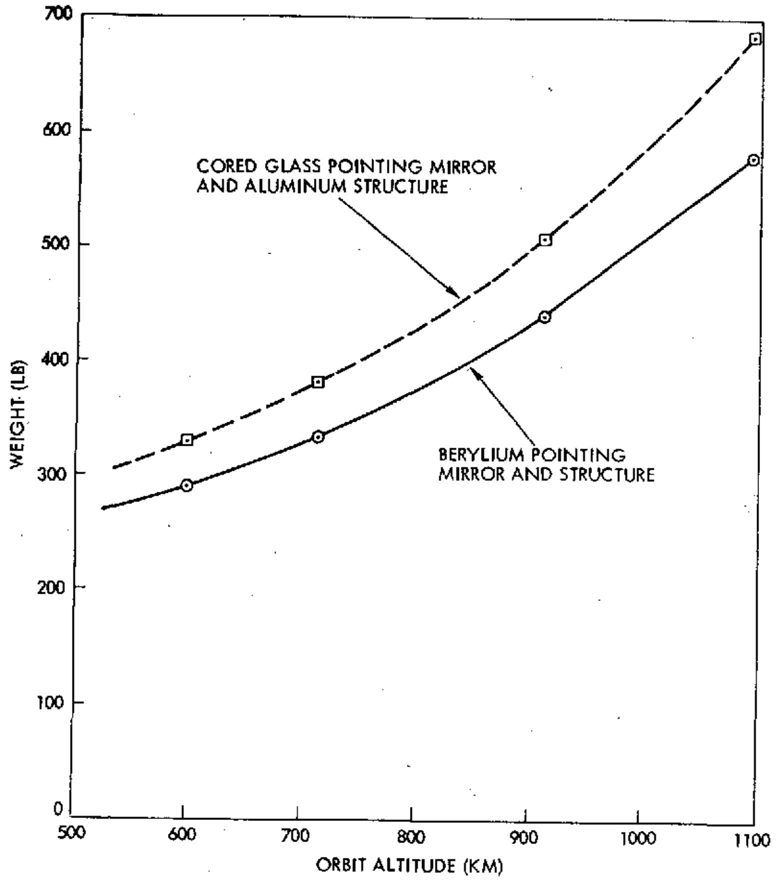


Figure 4-9. HRPI Weight Versus Orbit Altitude (Westinghouse Configuration)

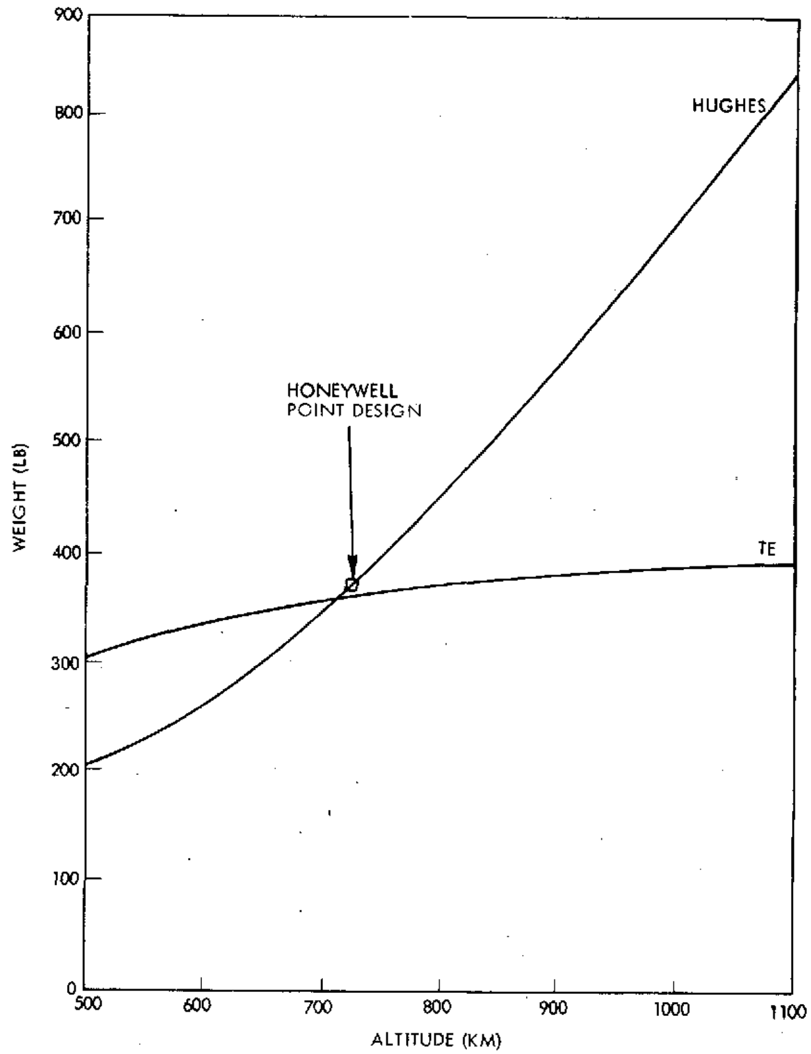


Figure 4-10. HRPI Weight Versus Altitude for Three Mechanical Scanners (Altitude Data Not Available for Honeywell Approach)

and mechanical scanners it must be emphasized that the SNR performance of the Westinghouse design is four times better than the mechanical approaches (see Reference 2 analysis). An electronic scanning HRPI with SNR equal to the mechanical scanners would weigh significantly less than the selected baseline instrument (the weights of the telescope and pointing mirror in Table 4-15 could be reduced by more than 50 percent).

Westinghouse estimates that the cost savings achievable by changing the design altitude from 914 to 717 km is 5 to 10 percent. This estimate can be compared with a NASA guideline, defined in Section 4.1, which relates cost to IFOV:

$$\text{Cost} \sim (\text{IFOV})^{-0.13}$$

For a given ground resolution, r, we have for the above altitude examples,

$$\frac{\text{Cost at 717 km}}{\text{Cost at 914 km}} = \frac{r/717}{r/914}^{-0.13}$$

$$= 0.97$$

The NASA guidelines therefore predicts a 3 percent savings. For cost analysis purposes, the Westinghouse 5 percent estimate would appear to be reasonable (2.5 percent cost savings per 100-km altitude reduction). Based upon a baseline cost estimate of $16 million (one refurbishable prototype plus one flight unit), this amounts to an estimated cost savings of $400 K per 100-km altitude reduction.

Significant weight savings can be achieved by using beryllium optical and structure components (Figure 4-9). Reduced payload weight may result in cost savings for other spacecraft components (e.g., launch vehicle, attitude control, and structure). These savings should be traded off against the additional cost of beryllium components. Westinghouse estimates that the additional cost for using beryllium would be less than 1 percent (<$160 K).

The HRPI cost estimate for mechanical scanning configurations is significantly higher than the electronically scanned detector-array design. At the baseline altitude (717 km) the estimated cost is $22 to $27 million (compared to $16 M for the Westinghouse design). The baseline cost and the cost/altitude dependence of the mechanical scanner HRPI's are essentially the same as the TM estimates (Table 4-12 and Figure 4-5). Relative cost as a function of altitude is given in Figure 4-11 for two mechanical scanner HRPI's (Hughes and Te) and the electronically scanned Westinghouse design.

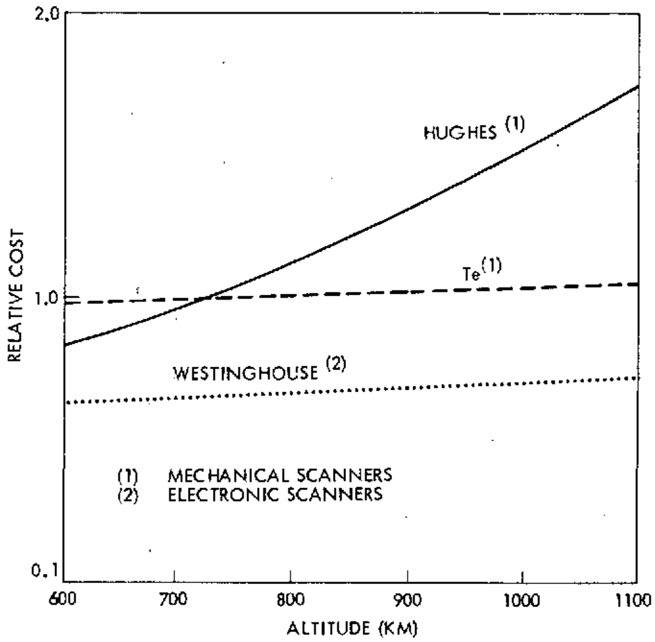


Figure 4-11. HRPI Manufacturers Cost Estimates as a Function of Altitude

4.3 OTHER INSTRUMENTS

Other missions may employ alternate instruments. It is of interest to consider how these sensors depend upon altitude. Three such payloads are dealt with.

4.3.1 Synthetic Aperture Radar

The operating power, weight, and data rate are only weakly dependent on orbit altitude. Although the average radiated power is theoretically directly proportional to altitude if all the other parameters, such as frequency, swathwidth, resolution, depression angle, and antenna length are fixed, in practice the other parameters may be varied to make the required power essentially constant over a wide range of altitudes.

If the operating power is unchanged, the weight will be affected only by the antenna size. The length of the antenna will be made as long as possible within the physical constraints imposed by the spacecraft. The height depends on the frequency and swathwidth as well as the altitude. If the swathwidth is fixed, then the reflector height will vary inversely with altitude. However, for the type of antenna proposed by Westinghouse, the weight of the reflector is only about 10 percent of the total synthetic aperture radar (SAR) weight so that a 2:1 variation of altitude would cause the SAR weight to vary about ± 5 percent.

Although the raw data rate is directly dependent on altitude, variations in the design parameters such as swathwidth, resolution, and depression angle with altitude will cause the raw data rate to vary. However, if the SAR operating power is fixed and it includes an output buffer (line stretcher), the output data will be essentially independent of altitude.

4.3.2 Passive Microwave Radiometer

With fixed-performance requirements such as swathwidth, spatial resolution, and temerature sensitivity, the main effects of altitude variation on the passive microwave radiometer (PMMR) are on the size of the antenna and the scan angle and rate.

The performance of a microwave radiometer is characterized by the following equation:

$$\Delta T \Delta W = K \left[\frac{Vh\phi}{\cot \theta} \right]^{1/2} \tag{35}$$

where

ΔW is the spatial resolution

ΔT is the temperature sensitivity

K is a constant dependent on the radiometer parameters

h is the altitude

ϕ is the scan angle

θ is the incidence angle of the radiometer beam

ΔV is the ground velocity of the radiometer beam.

For ocean missions, conical scan is required and the incidence angle will be fixed. Since the velocity is weakly dependent on the altitude, and ΔT and ΔW are assumed fixed, the above equation may be restated as:

$$\Delta T \Delta W \approx K \, (h \, \phi)^{1/2} \tag{36}$$

Defining K_o by

$$K_o = 2 \left(\frac{\Delta T \Delta W}{K}\right)^2 \tag{37}$$

yields

$$\phi = \frac{K_o}{h} \tag{38}$$

This result shows that for constant performance the scan angle and, therefore, the scan rate are inversely proportional to the altitude. In addition, the size of the antenna aperture required to maintain constant ground resolution is inversely proportional to the square foot of the altitude.

The effects of the scan rate and aperture size variations with altitude depend on the type of antenna used. If the planar array type of antenna is used, the area and, therefore the weight, will vary approximately inversely with altitude. If a mechanically scanned antenna is used the weight variations will be reduced, since only the reflector weight will vary with altitude. In addition, the antenna scan power will also vary inversely with altitude.

4.3.3 Five-Band MSS

The five-band MSS is well into development so that design optimization to alternate altitudes is probably inappropriate. However, it appears that the five-band MSS may be modified fairly readily to operate at any altitude between 250 and 500 nautical miles. As the altitude drops, the scanning mirror bumper position must be opened up and the image plane field stops increased in size. Hughes says both of these modifications are easy.

In addition, the mirror scan frequency will require some change due to the small change in spacecraft velocity, but this is readily accomplished by retuning the mirror.

Thus, it appears that a cost increment of some $0.5 million above the basic cost of $3.7 million for an "as-designed" scanner should be adequate to accommodate the necessary changes regardless of altitude. The basic cost was obtained from NASA's scanner cost equation. There will be no altitude impact on the weight of 140 pounds.

4.4 DATA COLLECTION SYSTEM

In general, any altitude increase in the range below 1000 nautical miles will be welcome, because:

- The permitted downlink EIRP will increase with no significant change in the maximum slant range to users. Overall system design (e.g., user ground stations) should be easier.

- The total available data collection system throughput will increase due to longer visibility periods a. higher altitudes.

4.5 WIDEBAND DATA HANDLING

The wideband data handling (WBDH) system is affected by mission altitude as shown in Table 4-16. The number of detectors in the baseline thematic mapper is increased as a function of altitude (refer to Section 4.1 for discussion of this phenomenon). The analog data multiplexers are, therefore, increased. The number of multimegabit operational data system (MODS) multiplexer mux boards varies from 7 to 10 depending on the orbit. The greatest change occurs in the speed buffer. The number of bits per swath is increased by a factor of 2.33 times, increasing the memory requirements a proportional amount.

The WBDH system weight and power can be estimated for other candidate thematic mappers at various altitudes by referring to Table 4-17 and Figure 4-12. Table 4-17 shows the cross-scan detectors per band as a function of altitude for various thematic mappers. These data are from existing designs. Weight-optimized designs will yield different numbers of detectors (see Section 4.1.1). Figure 4-12 shows how the WBDH system weight varies as a function of the number of cross-scan detectors/band. These exhibits permit evaluation of altitude-weight tradeoffs for alternate instruments.

Cost influences can be similarly developed. Fundamentally, the WBDH equipment costs will vary in proportion to the number of detectors (Table 4-17).

Neither weight or size will vary directly as a function of the launch vehicle selected.

4.6 IMPACT OF ORBIT ALTITUDE ON WIDEBAND COMMUNICATIONS SYSTEM

The wideband communications system (WBCS), contained within the WBCDH module, contains the RF equipment required to telemeter selected TM data, at 20 Mbit/sec, to low cost ground stations (LCGS) and full frame TM and HRPI data, at 256 Mbit/sec, to the wideband data collection stations. The 20 Mbit/sec selected data is transmitted in a

Table 4-16. Altitude Impact upon Wideband Data Handling System (Te TM; 15 Percent Bow-Tie Distortion) — Altitude Has No Effect on the HRPI Portion of the System

	Altitude (km)			
	600	717*	914	1100
Cross-scan detectors	12	16	29	32
Detector multiplexer channel	67	100	158	233
Spacecraft velocity factor	1.03	1.00	0.96	0.92
Data rate at over-sampling ratio 1.33 (Mbit/sec)	131	128	123	118
Over-sampling ratio to maintain 128 Mbit/sec	1.30	1.33	1.38	1.44
Quantity of boards				
MODS multiplexer (Te TM)	7	8	9	10
MODS controller	3	3	3	3
LCGS speed buffer	20	30	48	71
Other	5	6	8	10
Total	35	47	68	94
Weight (lb)/Power (w)				
Te TM module	6/11	7/12	7/12	8/12
HRPI module	9/13	9/13	9/13	9/13
WBWBCDH module	20/35	23/45	34/70	45/85
Total	35/59	39/70	50/95	58/110

*Baseline.

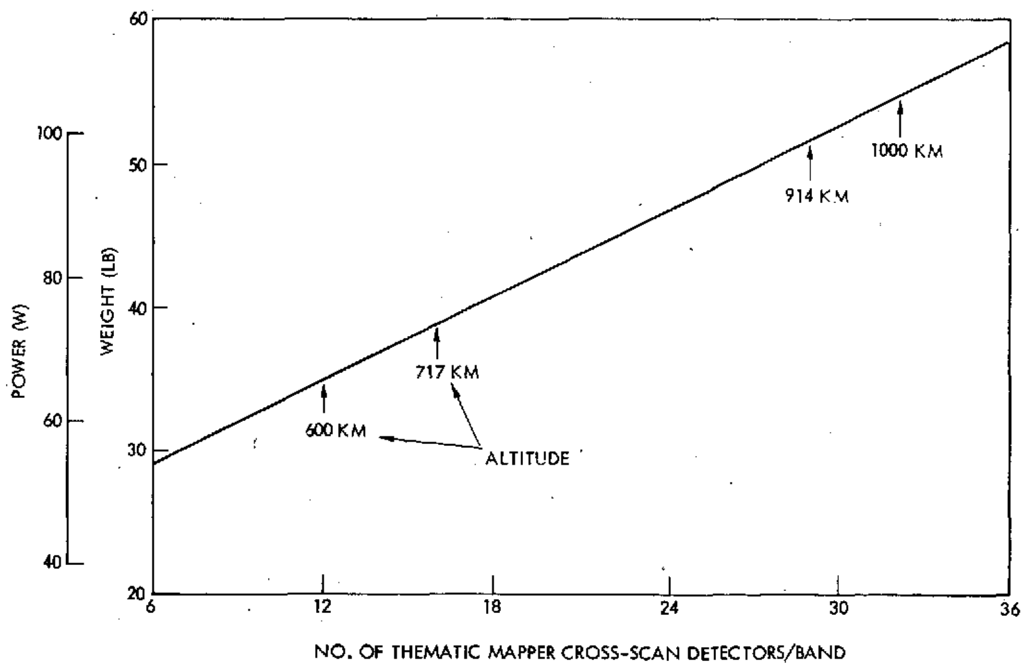


Figure 4-12. Total Weight and Power of the WBDH System as a Function of the TM Cross-Scan Detectors/Band

Table 4-17. Cross-Scan Detectors/Band as a Function of Altitude for Various Thematic Mappers

TM System	Altitude (km)			
	600	717*	914	1000
Te TM - SSR - breadboard	14	15	15	15
Te TM - SSR - optimized	14	16	21	25
Te TM - 15 percent bow-tie distortion	12	16	29	32
Hughes - cost optimized	6	8	14	18

*Baseline.

PCM, biphase-PSK format on an X-band carrier in the vicinity of 8 gHz. The 256-Mbit/sec data is transmitted in a quadriphase-PSK format also in the same frequency region.

The RF equipment contained within the module consists of a biphase modulator, a quadriphase modulator, two 0.5-watt power amplifiers, two pretransmission filters, and two gimballed X-band antennas.

The requirements for the EOS-A baseline design were established for a 716-km circular spacecraft orbit considering the impact of a NASA power flux density restriction which limited the incident RF power level at the earth's surface -152.0 dBw/m^2 in any 4-kHz band for angles of arrival between 0 and 5 degrees above the horizon and -142.0 dBw/m^2 in any 4-kHz band 90 degrees above the horizon (Reference 6). Table 4-18 presents a summary of the link calculations which were performed for the two downlinks. The power flux density for the 256 Mbit/sec link, at nadir is -152 dBw/m^2, while that for the 20 Mbit/sec link is -142 dBw/m^2.

6. "Radio Frequency Allocations for Space and Satellite Requirements," Mission and Data Operations Directorate, Goddard Space Flight Center, GSFC/M and DoD, 15 June 1973.

Table 4-18. EOS-A Baseline Wideband Communications Performance

Parameter	20 (Mbit/sec)	240 (Mbit/sec)	Comments
Transmitter power (dBw)	-3.0	-3.0	0.5 W
Transmit losses (dB)	3.5	3.5	-
Antenna gain (dB)	31.5	31.5	2-ft dish
EIRP (dBw)	25.0	25.0	-
RF path loss (dB)	178.6	178.6	2590-km slant range
Ground station antenna gain (dB)	44.5	54.5	-
Ground system temperature (^{o}K)	166.7^{o}K	204^{o}K	-
Performance margin (dB)	6.6	6.5	BER - 10^{-5}; 3 dB degradation

Table 4-19. Impact of Orbit Altitude

Orbit altitude (km)	716	1810	3706	36,041
5-degree slant range (km)	2590	4602	7276	41,384
Transmitter power (dBw)	-3.0 (0.5 w)	+2.0 (1.8 w)	+6.0 (4 w)	+13.0 (20 w)
Antenna gain - (dB)	31.5	31.5	31.5	31.5
Antenna diameter (ft)	2.0	2.0	2.0	5.0
RF path loss (dB)	178.6	183.6	187.6	202.7
Performance margin (dB)				
• 20 Mbit/sec	6.6	6.6	6.6	6.6
• 256 Mbit/sec	6.5	6.5	6.5	6.5

The impact of other spacecraft altitudes is presented in Table 4-19. In general, the impact of higher altitudes requires a higher spacecraft EIRP. Since the EOS-A baseline design requires an EIRP established by a 0.5-watt transmitter and a 2-foot dish antenna, higher orbits can be accommodated either by an increase in transmitter output power level, an increase in antenna gain (and, therefore, diameter), or a combination of the two. However, it is desirable to keep the antenna as small as possible from a gimballing, mounting, and pointing standpoint. If the antenna diameter is held constant at 2 feet, an 1810-km

(975 nautical miles) altitude can be accommodated with a 1.8-watt transmitter, while a 7276 km (3900 nautical miles) orbit requires a 6-watt transmitter. For synchronous operation, in order to keep the power amplifier at a reasonable level, in the vicinity of 20 watts, the 2-foot baseline dish would have to be replaced by a 5-foot one.

Design modularity dictates that sufficient flexibility be implemented into the baseline design to accommodate future missions or orbit changes with a minimum of change. Two approaches offer themselves with regard to accommodating sun-synchronous operation. The first is to simply replace the 0.5-watt power amplifier with the required higher powered unit. The other makes use of the 0.5-watt amplifier as a driver for a subsequent high power amplifier. In either case, the module design must allow enough room and secondary power to accommodate the higher powered unit.

Synchronous operation requires the use of a 20-watt (travelling wave tube (TWT) amplifier and a 5-foot dish. While the TWT can undoubtedly be handled within the module by providing for sufficient room and power, the larger antennas would require a different and more complicated mounting philosophy in order to preclude against mechanical interference. The preferred approach is to have one module design capable of handling all orbits (and missions) up to 1000 nautical miles (1850 km) and another to handle the synchronous case.

Operation in the 300 to 900 nautical mile range will have no significant impact upon the size, weight, or cost of wideband communications, unless other parameters change (e.g., due to a major change in payload data rate). Going to geostationary operation would be a major impact for these data rates. However, the payloads under consideration for synchronous application will have a more significant wideband communication impact than will the altitude.

4.7 PAYLOAD STRUCTURE AND THERMAL

4.7.1 Structure

The character of the payload structure depends primarily upon the composition of the payload and the launch vehicle. In the orbit altitude range of interest, orbit selection will have no direct effect upon the payload structural design.

Launch vehicle selection can affect the payload structural concept and design significantly, because this choice interacts with the decision on whether to provide for on-orbit resupply of the instruments. In the Titan-launched configuration, each instrument (and other payload element) is an individual module with its own structure. These modules then fit within payload bays in a payload structure, which is a mission-peculiar.

Use of a Thor-Delta 2910 requires close attention to minimizing structural weight. Therefore, the recommended approach is to delete the payload module structures and attach the instruments directly to the mission-peculiar payload structure (Figure 4-13). In either of these cases, benign thermal conditions will be achieved by thermal isolation of the payload elements from the structure.

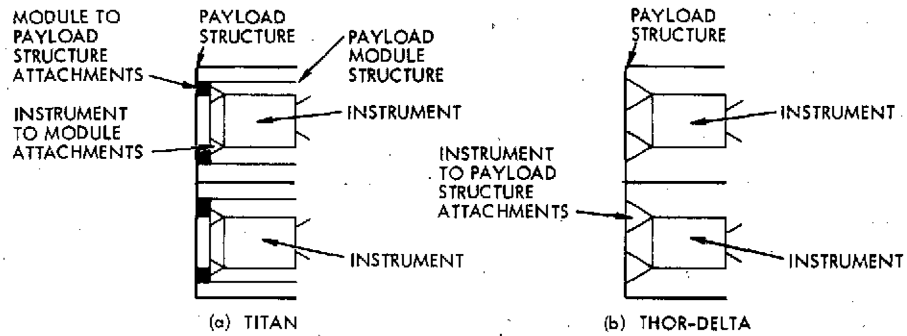


Figure 4-13. Instrument-to-Payload Structured Interface

4.7.2 Thermal

Payload thermal control is based on the concept of thermally independent modules with module/structure conductive interaction minimized by controlling both mechanical coupling and temperature gradient across the interface. Radiative interaction is minimized by use of multilayer insulation (MLI) between module and structure. Each module has a thermal control system consisting of a temperature controlled heater circuit and passive elements of MLI and low α_s/ϵ radiator coating. Module thermal control constraints include module/structure interface requirements. Payload structural frame temperature level, distribution, and fluctuations are controlled with use of several independent heater circuits optimally located on the frame, and MLI blankets sandwiching structural members. Details of the structural thermal control system and of the module thermal control design are presented in Reference 1.

The baseline thermal control system is designed for a near-earth 11:00 a.m. sun-synchronous orbit, with heater power level and radiator area sized for this case. For other node/sun phasing from 6:00 a.m. to 12:00 noon sun-synchronous orbits, heater power level and radiator area can be resized; however, the reference design can be used for all node/sun phase orbits with a heater power penalty. This heater power penalty will also apply to non-sun-synchronous orbits where phasing between 12:00 noon and 6:00 p.m. is accommodated by a 180 degree yaw turn.

The effects of a geostationary orbit upon payload structure thermal control cannot be assessed in general, since they will depend on the detailed payload configuration and the spacecraft orbital configuration. The thermal design for the payload structure will be a mission-peculiar; however, the concept defined here will apply.

The influence of launch vehicle selection upon thermal design for the payload structure relates to the question of on-orbit resupply, as discussed in Section 4.7.1. Without the need to resupply, thermal insulation on the Thor-Delta configuration can be custom installed to minimize heat leaks, resulting in less thermal design uncertainty than for the Titan payload structure, resulting in a lower heater power allocation. Moreover, for the Thor-Delta spacecraft the inner structural MLI blanket will not be required.

Thermal costs for the Thor-Delta payload can be expected to be less than for the Titan payload but this cost differential will not be appreciable.

5. SPACECRAFT DESIGN CONSIDERATIONS

A keynote in development of the basic modular spacecraft design has been securing a high degree of insensitivity to mission, launch vehicle, and orbit. To this end the spacecraft has been partitioned into general-purpose and mission-peculiar elements:

- General-purpose
  - Communication and data handling
  - Attitude determination
  - Spacecraft structure
- Mission-peculiar
  - Electric power
  - Actuation (control and propulsion)
  - Solar array and drive.

The general-purpose modules are generally independent of mission, orbit, and launch vehicle.* The mission-peculiar elements have a direct and primary dependence upon mission, orbit, and/or launch vehicle. The remainder of this section delineates the orbit and launch vehicle influences on each item.

5.1 COMMUNICATIONS AND DATA HANDLING MODULE

The EOS-A communications and data handling (CDH) module baseline design was determined by considering the requirements by an orbit between 300 and 900 nautical miles. Operation at higher orbits generally will have a direct impact on the communication system and no impact on the data handling system. This section discusses the impact of orbit altitude on the performance, system design, and cost of the CDH module.

Table 5-1 summarizes the link performance margins calculated for the S-band uplink and downlink at the baseline altitude (716 km). The baseline design uses two omni-directional antennas suitably combined to

*There can be a secondary dependence in extreme cases (an added altitude sensor, increased transmitter power, additional battery capacity, etc.).

achieve spherical coverage for both reception and transmission. As shown in the table, very high uplink carrier and command performance margins are obtained through the use of the normal NASA STDN 9-meter dish and 1-kw transmitter output level. Downlink performance is adequate with the use of omni-antennas and a 2-watt transmitter output power level. It should be noted that there are two operational modes for downlink telemetry. In mode 1, real-time housekeeping and computer memory dump data at 32 kbit/sec is transmitted in frequency division multiplex form with 512 kbit/sec medium-rate user data. In mode 2, the medium-rate data is replaced by transponder ranging data.

Table 5-1. EOS-A Baseline Communications Performance

Parameter	Worst Case Value	Notes
Uplink		
Frequency (MHz)	2050.0	-----
Ground station EIRP (dBm)	103.0	STDN 9-meter dish
RF path loss (dB)	166.9	716-km orbit, 2590 km slant range
Spacecraft antenna gain (dBi)	-3.0	-----
Performance margins:		
• Carrier (dB)	49.2	800 Hz loop
• Command (dB)	47.3	10^{-6} Bit error rate
Downlink		
Frequency (MHz)	2226.2	
Spacecraft transmitter power (dBm)	33.0	2 watts
Spacecraft antenna gain (dBi)	-3.0	
RF path loss (dB)	167.6	716 km orbit, 2590 km slant range
Mode 1 performance margins		
• Carrier (dB)	34.6	800 Hz loop
• 32 kbit/sec telemetry (dB)	5.1	10^{-6} Bit error rate
• 512 kbit/sec telemetry (dB)	8.3	
Mode 2 performance margins		
• Carrier (dB)	37.1	800 Hz loop
• 32 kbit/sec telemetry (dB)	17.4	10^{-6} Bit error rate
• 500 kHz tone ranging (dB)	35.9	

Table 5-2 shows the impact of orbit altitude on the CDH communication system. As is indicated, acceptable telemetry performance (+3 dB margin) can be maintained through the use of the baseline omni-antenna and 2-watt transmitter up to an orbit altitude of 1810 km (977 nautical miles). Beyond this point, out to about 3706 km (2000 nautical miles), acceptable performance can be achieved through the use of an omni-antenna providing +1.7 dBi gain and the baseline 2 watt transmitter. At synchronous altitude (36,041 km), a nonpointable, 2-foot parabolic antenna with a 2-watt transmitter will be required. Finally, more than acceptable performance is obtained on the uplink out to synchronous altitude with the use of the baseline omni antenna.

Table 5-2. Impact of Orbit Altitude on Communication System

Orbit altitude (km)	716	1810	3706	36,041
5-deg slant range (km)	2590	4602	7276	41,384
Downlink (2226.2 MHz)				
RF path loss (dB)	167.6	171.9	176.6	191.7
Transmitter power (w)	2.0	2.0	2.0	2.0
Antenna type	omni	omni	omni	omni
Antenna diameter (ft)	--	--	--	--
Antenna gain (dB)	-3.0	-3.0	+1.7	29.5
Antenna beamwidth (deg)	220	220	80.0	17.0
Mode 1 performance				
• Carrier (dB)	33.8	29.5	29.5	32.2
• 32 kbit/sec telemetry (dB)	7.3	3.0	3.0	5.7
• 512 kbit/sec telemetry (dB)	8.4	4.1	4.1	6.8
Mode 2 performance				
• Carrier (dB)	41.9	37.6	37.6	40.3
• 32 kbit/sec telemetry (dB)	20.0	15.7	15.7	18.4
• 500 KHz ranging (dB)	18.3	14.0	14.0	16.7
Uplink (2050.0 MHz)				
RF path loss (dB)	166.9	171.2	175.9	189.8
Antenna type	omni	omni	omni	omni
Antenna gain (dBi)	-3.0	-3.0	-3.0	-3.0
Performance				
Carrier (dB)	49.2	47.1	42.4	28.5
Command (dB)	47.3	45.2	40.5	26.6

An additional comment relates to the software which resides within the on-board computer. The only changes in the software will be in parameter values over the range 300 to 900 nautical miles. Changes at higher altitudes will be more payload related than orbit related.

5.2 POWER MODULE

5.2.1 Launch Vehicle Dependence

The power module is functionally independent of the launch vehicle and no modifications of the electrical system are required. Minor mechanical modifications are required to provide direct fastening of the module to the structure for the non-resuppliable Thor-Delta missions.

5.2.2 Orbit Dependence

The period of battery discharge (eclipse) increases slowly with orbital altitude for sun-synchronous high noon orbits. For orbits for which the orbit normal does not make a right angle with the local sun vector, the battery discharge period diminishes at high altitudes. Regardless of orientation the total orbit period and frequency of discharge decreases with increasing altitude. Both frequency of discharge and duration (depth) of discharge impact battery size for any fixed-life requirement. The effect of altitude on allowable depth of discharge for a nominal 11:00 a.m. orbit is illustrated in Figure 5-1. Battery sizing is determined from the load requirements, duration of discharge, and allowable depth of discharge. Total ampere-hour requirements of 1.0- and 0.5-kw loads are shown in Figure 5-2 for a range of altitudes.

The effect of operating at geosynchronous altitude is not consistent with this parametric study. The low frequency of discharge at this altitude will allow approximately a 60 percent depth of discharge. Further evaluation of the geosynchronous mission must await better definition of the load profile since the potentially low rate of charge available from an optimized array and an assumed constant load may require modifications of the charge mode. This problem arises because the efficiency of recharge drops off rapidly at low charge rates. Potential solutions include computer sequencing of batteries and loads to decrease the duty cycle and increase the efficiency of battery recharge.

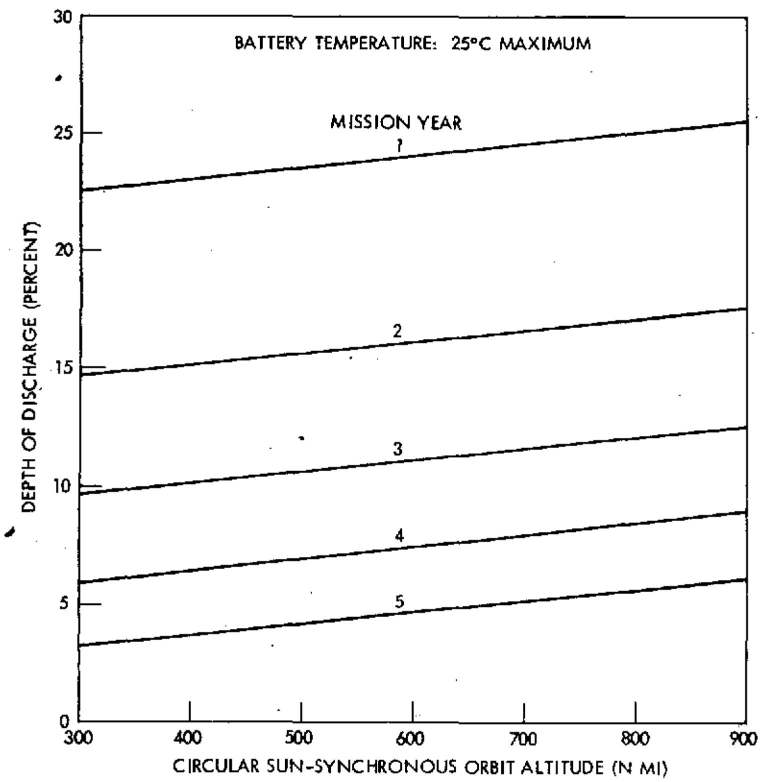


Figure 5-1. Maximum Battery Depth of Discharge (Percent) Based upon Rated Cell Capacity

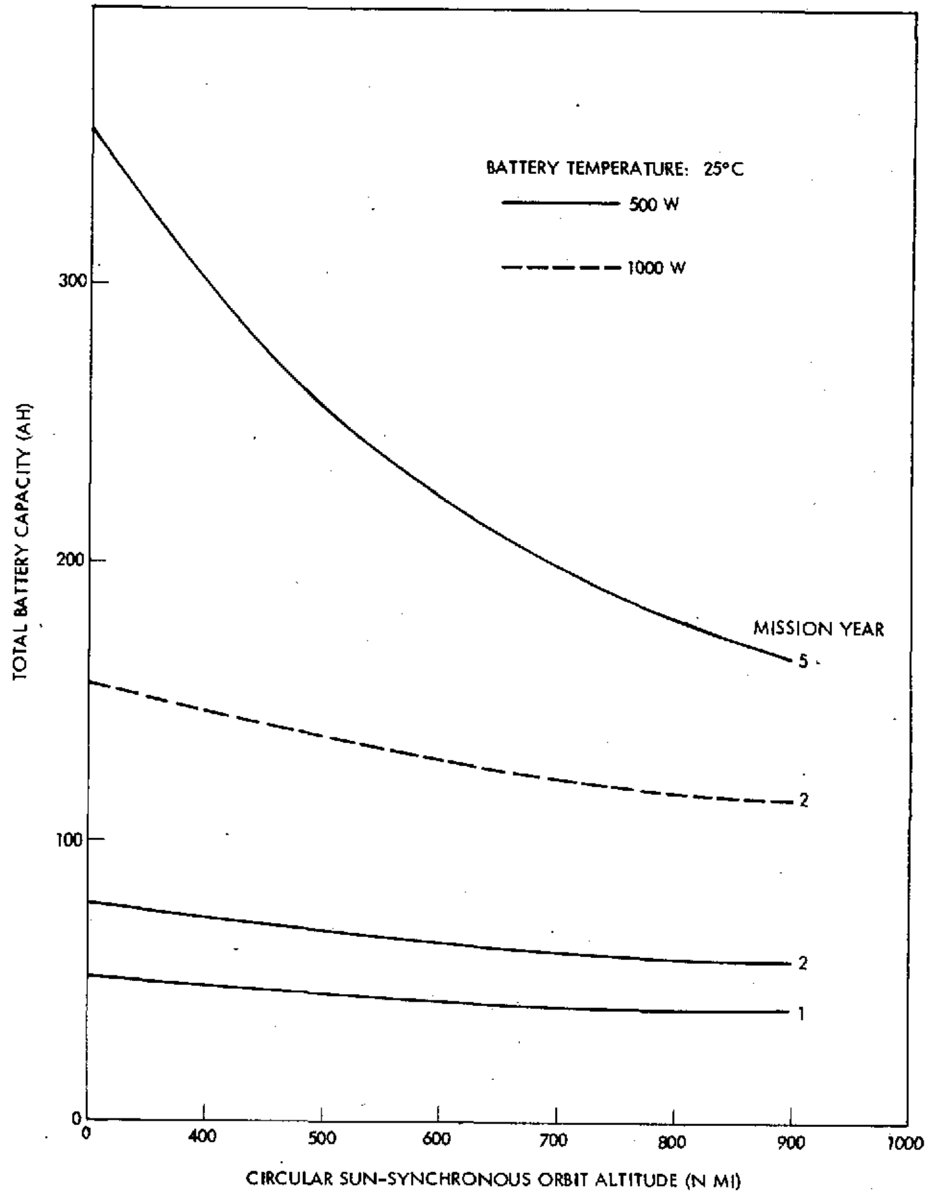


Figure 5-2. Average Occult Discharge Power

Variations of the local time of the ascending node of sun-synchronous orbits influence the duration of the eclipse period with consequent impact on battery sizing. The local time of non-sun-synchronous orbits varies continuously from high noon orbits to twilight orbits. The battery requirements on all of these orbits will be less than for the nominal EOS-A orbit assuming equivalent loads. Smaller batteries may be used for these missions, however it is doubtful that the design, qualification, and associated documentary controls for several battery designs is worth the saving on unit cost.

Temperature control of the battery is a critical power module requirement. Thermal constants are influenced by variations of sun-synchronous orbit local time due to effects of the angle of solar incidence and albedo and the temperature of the earth. The location of the power module on the +Y axis of the vehicle minimizes the effects of direct solar heating. The effects of orbit local time of the thermal characteristics of the power module are illustrated in Table 5-3.

Table 5-3. Thermal Effects of Orbit Local Time on Power Module

Orbit Local Time	Heat Rejection Capability (w/ft^2)	Altitude (n mi)
6:00 a. m.	30. 0	487
9:00 a. m.	27. 8	487
12:00 a. m.	27. 6	487
10:30 a. m.	27. 3	300
10:30 a. m.	28. 3	900
Geosynchronous	22. 6	19, 363

It is obvious that the basic EOS-A power module design can perform satisfactorily for all potential orbits.

5.3 ATTITUDE DETERMINATION MODULE

The attitude determination module incorporates the non-mission-peculiar attitude control sensors, inertial reference assemblies, and associated electronics. The attitude determination module is of a fixed-design independent of launch vehicle or orbit selection. Therefore, design and cost of this module are independent of launch vehicle or orbit selection.

5.4 ACTUATION MODULE

The actuation module contains the following subsystems:

- Hydrazine propulsion for orbit injection and velocity trim
- Cold gas nitrogen reaction control thrusters for large angle reorientations and attitude control during hydrazine engine burn and for use during attitude control failure modes
- Normal mode attitude control torquers (reaction wheels and magnetic torquers)
- Drive electronics for reaction wheels, magnetic torquers, and thruster valves.

The basic spacecraft structural arrangement (four-sided versus triangular) affects the configuration of the actuation module structure (see Reference 1). The impact of launch vehicle and orbit selection on the elements within the actuation module are discussed in the following paragraphs.

Actuator sizing, particularly for wheels and magnetic torquers, depends strongly upon disturbance torque magnitude. Figure 5-3 shows four bounding models which have been used in selecting an inventory of standard actuators (Reference 1). These curves clearly show the dependence of the dominant disturbance torques on orbital altitude. These torques will also depend upon launch vehicle, to the extent that the launch vehicle constrains the configuration design.

5.4.1 <u>Reaction Wheels</u>

Reaction wheel size (momentum storage) requirements are a strong function of orbital altitude and the associated disturbance torques (Figure 5-4). Also impacting reaction wheel sizing is the spacecraft attitude requirements as to earth orientation or inertial orientation. The actuation module is designed to accommodate a family of standard reaction wheel designs from which members can be selected to cover the spectrum of EOS mission requirements. Therefore, the cost of this element is essentially independent of launch vehicle or orbit selection. Of course, weight will vary with the standard unit selected.

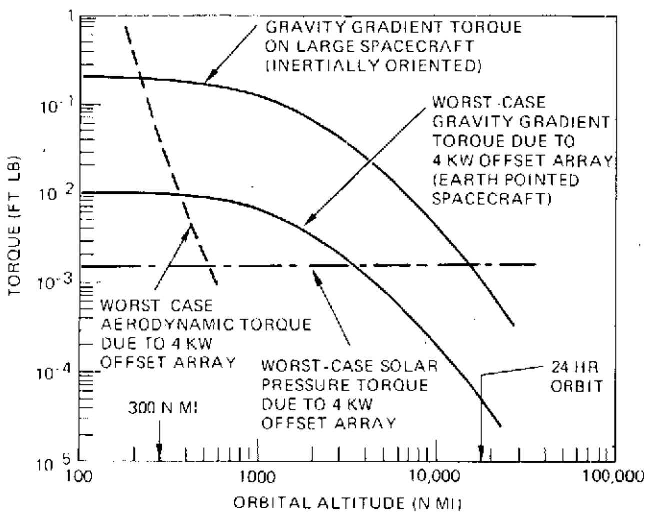


Figure 5-3. Disturbance Torque Trends

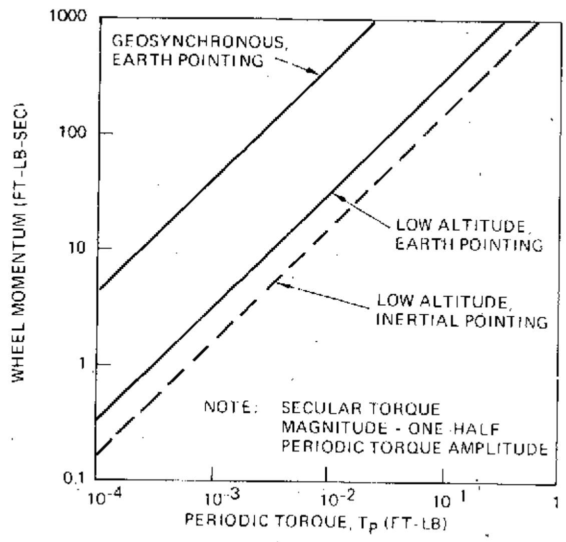


Figure 5-4. Wheel Momentum Requirement (Theoretical)

5.4.2 Magnetic Torquers

The magnetic torquer field requirement is a strong function of orbital altitude and the associated earth magnetic field and disturbance torques. Also impacting magnetic torquers sizing via the disturbance torques is the spacecraft attitude requirement as to earth orientation or inertial orientation. The actuation module is designed to accommodate a family of standard length magnetic torquers from which can be selected units compatible with any specific EOS mission. The cost of this element will be essentially independent of orbit or launch vehicle selection. Weight will increase with magnetic moment. Figure 5-5 shows the characteristic dependence of magnetic moment upon disturbance torque magnitude and weight variations are represented in Figure 5-6.

5.4.3 Actuation Module Electronics

The module electronics which provide control of the reaction wheels, magnetic torquers, and thrusters are of standard design compatible with extreme mission requirements. Therefore this element is mission non-peculiar and its cost is independent of launch vehicle and orbit selection.

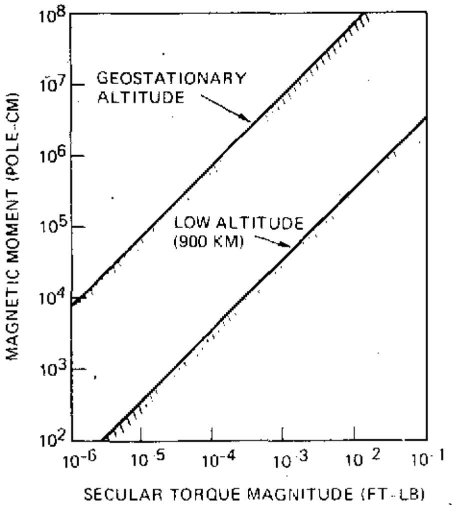


Figure 5-5. Minimum Magnetic Moment Requirements

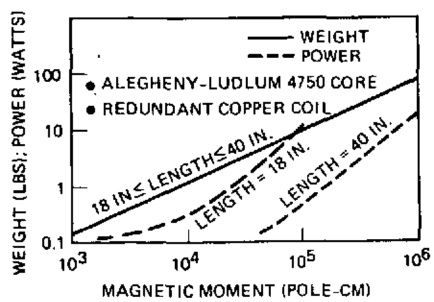


Figure 5-6. Magnetic Torquer Sizing

5.4.4 Mass Expulsion Systems

The mass expulsion systems are mission-peculiar, and depend on both the orbit and launch vehicle selection. This dependence primarily affects the hydrazine velocity adjustment system, with small related changes in the cold gas (nitrogen) system. The major dependencies are:

- Orbit Selection. Aerodynamic drag and the quantity of propellant required for drag makeup depend strongly upon orbital altitude as shown in Figure 3-12.
- Launch Vehicle Selection. Of the launch vehicles selected, only the Thor-Delta favors direct injection. In other cases, an integral circularization capability will be included in the actuation module, with significant quantities of hydrazine required (Figure 3-14) and a relatively high-thrust level implied (e.g., 50 pounds).

The baseline designs described in Appendix A to Report No. 3 reflect these parametric influences. The propulsion system for the Titan launched EOS is approximately 20 percent more expensive than for a Thor-Delta launch. The design (and cost) of the propulsion system is essentially independent of orbit selection because the propellant tanks have been sized for a worst-case requirement and will be off-loaded for less stringent missions.

5.5 SPACECRAFT STRUCTURE AND THERMAL

5.5.1 Structure

Each subsystem module will have an independent module structure. The basic module structures will be the same for both in-orbit serviceable and nonserviceable spacecraft; in the latter only those mechanisms associated with in-orbit serviceability will be deleted. In either case, the thermal interface between modules and spacecraft structure will be the same. The module structures will not contribute to the stiffness of the basic spacecraft structure, due to the manner of attaching them.

The basic structural configuration (module arrangement) can vary with launch vehicle, but will be orbit independent. The Titan vehicles lend themselves to a four-sided module arrangement which makes the most efficient use of the fairing dimensions (Reference 1). In the Thor-Delta configuration, the modules are arranged triangularly, with the actuation module at the aft end. Note, however, that the Thor-Delta version is compatible also with the Titan and the Shuttle and is, therefore, a universally applicable configuration. The decision regarding providing the capability for on-orbit servicing (which adds roughly 380 pounds to the Thor-Delta configuration) will ultimately depend upon the total Observatory weight for a particular mission and the orbit selection constraints.

5.5.2 Thermal

Spacecraft thermal design dependence upon orbit selection and launch vehicle is similar to that of the payload section, described in Section 4.7.2.

5.6 SOLAR ARRAY AND DRIVE MODULE

The design of the array and drive module is affected by orbital altitude and phasing, and, in a secondary manner, by launch vehicle selection.

5.6.1 Altitude Effects

All other factors being invariant, the altitude selected will affect array sizing via radiation-induced degradation over the selected lifetime.

Table 5-4 shows an example of this influence, for two coverglass thicknesses. In the low-altitude range, array performance degrades monotonically with altitude with the effect more pronounced with the thinner coverglass. This effect is compensated slightly by the decrease in the proportion of time spent in eclipse but this improvement will be minor.

Table 5-4. Array Output (w/ft^2) After 2 Years in Orbit

Altitude (n mi)	Output (w/ft^2)	
	6-mil cover	20-mil cover
300	7.37	7.52
387	7.03	7.41
450	6.79	7.30
600	6.17	6.90
800	5.54	6.22
1000	5.13	5.84
19,300 (24^h)	5.99	6.77

The data presented above could lead to selection of the 20-mil configuration where weight is noncritical, primarily because this configuration can prove to be less expensive (e.g., less breakage, etc.). However, the alternate array offers lower weight (e.g., for a 950-watt output at 2 years, a 17.5 pound increment at 387 nautical miles) and, therefore, more payload capacity.

5.6.2 Orbit/Sun Phasing

Orbit/sun phasing affects the solar array and drive module materially. It is most useful to consider non-sun-synchronized orbits, since they include as "snapshots" all possible sun-synchronous orbit phasings.

For low-altitude nonsynchronous near-polar orbits, the angle made by the sunline with the orbit plane will take on all possible values throughout some fraction of a year, as both the sunline and the orbit nodal line move in inertial space. To maintain optimum power conversion, a two-axis array drive would be required (e.g., a continuous orbit-rate-drive about the pitch axis, plus an intermittent hinge drive about an orthogonal axis). Note that in order to avoid sun interference with the star sensors in the attitude determination module (and to avoid solar impingement on radiations) the spacecraft will effect a 180-degree yaw turn whenever the sunline passes through the orbit plane (noon condition). This turn will benefit the array module design, avoiding array shadowing by the spacecraft and allowing the hinge freedom to be only 90 degrees.

If the orbit considered is sun-synchronous, the hinge drive mentioned above can be replaced by a fixed-angle hinge tailored to the orbit plane angle (e.g., 15 degrees for the baseline 11 a.m. orbit). This approach of a fixed-hinge may also have merit for a nonsynchronous orbit, in which case the hinge angle would be optimized (considering eclipse characteristics, etc.). This concept is discussed in Reference 1, Appendix A, Section 5.4.3.

In general, the solar array and drive module can be readily implemented for any orbit and is, therefore, not a factor which will inhibit free orbit selection.

6. GROUND SYSTEMS

6.1 LOW COST GROUND STATION

The choice of the launch vehicle and orbit does not have a significant effect on the design or cost of the implementation of the LCGS. This conclusion assumed that none of the candidate sensors will be excluded because of launch vehicle considerations, and, additionally, that orbital control of altitude variations and cross-track drift will be the same for any orbit selection in the class of low-altitude sun-synchronous, circular orbits of interest, and that the same sensor ground resolution is used.

The orbit selection could affect the tracking strategy and sensor data rates (hence, LCGS processing configurations). For a predicted ephemeris accuracy of 700-m (3σ), the ground antenna tracking error due to orbital uncertainty will be less than 0.1 degree (3σ) for any orbit selection in the class specified, and will not preclude the lower cost programmed tracking design approach. Furthermore, the data rates will remain within 6 percent of the rates determined for the 716-km baseline orbit and, therefore, will not affect the basic processing configurations of the LCGS. It is also to be noted that changes in data quality due to the changes in ground speed and viewing aspect will be insignificant relative to the baseline orbit case.

6.2 CONTROL CENTER AND RELATED FACILITIES

In any control center design concept for spacecraft in relatively near-earth orbits, the driving functions for the design are the spacecraft's orbital period, the amount of payload operation and contact with ground telemetry, tracking and command, or payload readout stations.

6.2.1 Operational Control Center

A baseline orbit of 384 nautical miles was used for timeline analysis. However, none of the results or concepts were critically dependent on either spacecraft altitude or period, at least in the assumed ranges. If a very low-altitude orbit were used, the control center cycles, which are geared to real-time passes, must speed up a few minutes but the actual period change from baseline is almost inconsequential. If a maximum altitude is assumed, the principal impacts are twofold. The control center cycles are longer but the HRPI scheduling becomes more troublesome as the HRPI will be able to scan target areas much further away from the ground track than is possible at lower altitudes. This results in doubling or tripling the output of the area availability model. The HRPI scheduling has no effect on the control center hardware because a large margin of processor time and lesser margin of memory were allocated for this type of potential growth in requirements. The increase in data base due to target availability has no effect on the software design for the control center.

The timelines for the baseline mission had a 2-hour contingency reserve. If the HRPI scheduling requirements were increased by large amounts, multiple shift operations could be required in mission scheduling.

6.2.2 STDN Stations

Acquisition ranges, real-time telemetry data, payload data, and time for command transmission are all a function of orbit altitude. None are so sensitive in the ranges assumed as to compromise operations. If a high-circular orbit is assumed, the Alaskan station particularly, will be able to receive much more real-time telemetry than with lower orbits and receive at least some indication of vehicle health on every orbital pass.

6.2.3 NASCOM Network

The operation of the NASCOM network is relatively insensitive to orbital altitude in the ranges assumed. With higher altitudes, the telemetry data load obviously increases but the anticipated increase should pose no critical NASCOM problem.

6.3 CENTRAL DATA PROCESSING FACILITY

Under the same assumptions as stated for the LCGS (i.e., no candidate sensor exclusion because of launch vehicle considerations, same orbital control for each orbital selection, and same ground resolution for sensor data), the choice of the launch vehicle and orbit does not have a significant effect on the design or cost of the implementation of the central data processing facility (CDPF) with one exception. If, to minimize bowtie distortion as altitude increases, the number of detectors in the thematic mapper is increased (Section 4. 1) the CDPF will be affected because the number of detectors determines the scan volume stored to reformat the data into line-sequential data. The cost of the memory to store this volume in the reformatting system thus will increase with altitude. Figure 6-1 presents the cost impact on the reformatting memory as a function of the number of detectors per band for bands 1 through 6; band 7 has 1/4 the number of bands as the others. The number of detectors, as a function of altitude and thematic mapper design, is presented in Section 4. 1

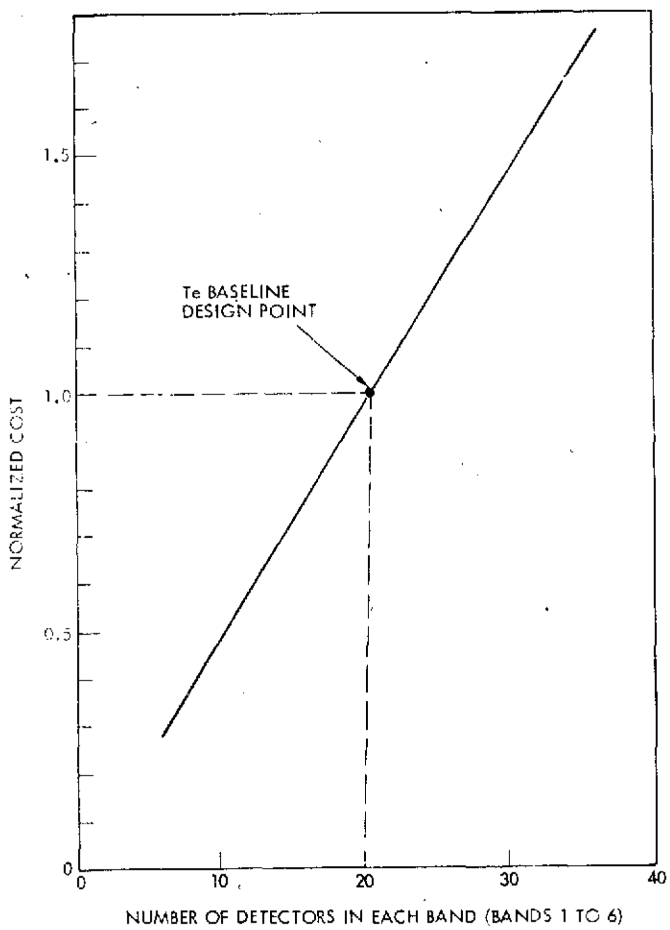


Figure 6-1. Normalized Cost Impact on Reformatting Memory by Number of Thematic Mapper Detectors in Each Band of 1 to 6

The orbit selection affects the sensor data rates and volumes. As stated before, the data rates will remain within 6 percent of the rates determined within the same limit. These changes do not affect the basic processing configuration or archiving volume considerations. Data quality will also be affected by changes in the ground speed and viewing aspect induced by altitude changes. When referenced to the baseline 717-km orbit, the data quality with respect to design and cost will be insensitive to changes in orbital altitude within the class of orbits considered.

7. CONCLUSIONS AND RECOMMENDATIONS

The preceding sections have presented the factors which influence orbit and launch vehicle selection for EOS-A and related low-altitude LRM missions. Specific recommendations require specific missions to be addressed. However, general conclusions can be drawn and later particularized to the missions (R and D and operational) identified in Section 2.1.

The cost element which depends most strongly upon orbit/launch vehicle selection is the cost of the launch vehicle itself. Other elements tend to depend only weakly upon these selections. Therefore, interrelationships among Observatory weight, orbit altitude, and launch vehicle capability are the dominant consideration.

Table 7-1 indicates the total launch vehicle payload (including adapter) and the maximum attainable altitude (with no contingency allowance) using Thor-Delta vehicles, with capabilities as shown in Figure 7-1, for a variety of mission and spacecraft design options.* Based on design studies presented in Reference 1, four redundancy levels, defined in Table 7-2, are included; for an infinite design life, mean mission duration (MMD) is identical to mean time to failure (MTTF) and can be employed in an analgous manner in preliminary mission cycle costing analyses. This figure, with the launch vehicle costs of Table 3-6 and the orbit analyses of Section 3.1, can be used as a data base to develop specific recommendations.

7.1 EOS-A MISSION (TM PLUS HRPI PAYLOAD)

At the outset of the study, two specific baseline orbits were established, based on proposal studies (Table 7-3). A low-altitude Thor-Delta orbit was selected, due to the limited capability of that launch vehicle in the 2910 version. The higher altitude orbit for the Titan version reflects

*It is worth noting that, in the altitude range of primary interest (e.g., 300 to 500 nautical miles), allowing 100 pounds of contingency will reduce the reachable altitude by about 50 nautical miles.

Table 7-1. Launch Vehicle Payloads and Altitude Limits (No Contingency)

Mission	Configuration	Resupply	Launch Vehicle Payload (lb)				Maximum Altitude, nmi (2910/3910)			
			Minimum	Variant 1	Variant 2	Nominal	Minimum	Variant 1	Variant 2	Nominal
EOS-A (R and D)	Quadrangular; TM + HRPI	Yes No	4918 4308	4974 4364	4993 4383	5030 4420	NA NA	NA NA	NA NA	NA NA
EOS-A (R and D)	Triangular; TM + HRPI	Yes No	3089 2604	3145 2660	3164 2679	3201 2716	170/650 410/920	150/620 380/890	140/610 370/880	120/600 350/860
EROS (Operational)	Triangular; 1-MSS	Yes No	2236 1948	2292 2004	2311 2023	2348 2060	630/** 820/**	590/** 790/**	580/** 770/**	550/** 750/**
EROS (Operational)	Triangular; 2-MSS	Yes No	2694 2315	2750 2371	2769 2390	2807 2427	360/870 575/**	330/840 540/**	320/830 530/**	300/800 510/**
EROS (Operational, R and D)	Triangular; 1-MSS + TM	Yes No	2933 2506	2989 2562	3008 2581	3045 2618	240/740 460/980	220/710 430/950	210/700 420/940	190/680 400/920
EROS (Operational, R and D)	Triangular; 2-MSS + TM	Yes No	3370 2855	3426 2911	3445 2930	3482 2967	*/510 280/780	*/490 250/750	*/480 240/740	*/460 220/720
EROS (Operational, R and D)	Triangular: 1-MSS + HRPI	Yes No	2903 2476	2959 2532	2978 2551	3015 2588	260/760 480/1000	230/720 440/960	220/710 430/950	200/690 410/930
EROS (Operational, R and D)	Triangular 2-MSS + HRPI	Yes No	3340 2825	3396 2881	3415 2900	3452 2937	*/530 290/800	*/500 270/770	*/490 260/760	*/470 240/730

Notes: NA — not applicable
* — altitude below 100 n mi
** — altitude above 1000 n mi

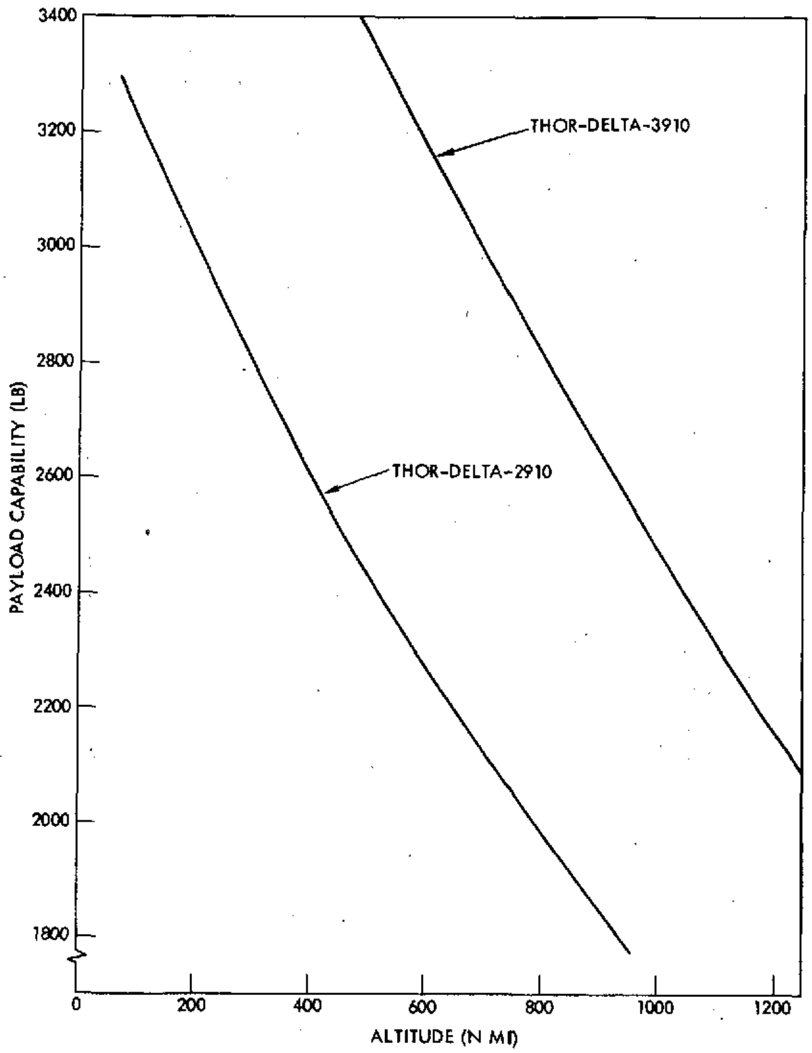


Figure 7-1. Payload Capability to Circular Sun-Synchronous Orbits (Direct Injection)

the desire to use the increase in payload capability to achieve more ground station coverage and lower aerodynamic drag (hence, less frequent orbit adjust).

During the study, these conclusions were reassessed. A major factor in this review process was the frequency of HRPI repeat coverage (Table 3-1). This analysis showed that more favorable repeat characteristics can be achieved at slightly different altitudes (Table 7-4). Notice that these altitudes are consistent with launch vehicle capabilities (Table 7-1), if an unresuppliable Thor-Delta 2910 is assumed.* Considering the sizeable cost differential between the Thor-Delta 2910 and the Titan IIIB, a Delta launch to the lower altitude of Table 7-4 is recommended for EOS-A.

7.2 OPERATIONAL LRM MISSIONS

A wide range of operational LRM missions can be defined. Two such missions were suggested by NASA/GSFC personnel in recent meetings (Section 2.1). Table 7-1 treats each of these, with the R and D

*Titan weight contingencies are 100 to 200 pounds in the resuppliable case and 700 to 800 pounds in the unresuppliable case (see Reference 1).

Table 7-2. Redundancy Options

Redundancy Option	Description
Minimum	Minimum redundancy necessary to ensure no single failure preventing retrieval/resupply. Spacecraft MMD is about 9 months (3-year design life), not including payload
Variant 1	Limited added redundancy, giving MMD of about 17 months
Variant 2	Still more redundancy, giving MMD of about 22 months
Nominal	Most electronics made standby redundant; "typical" redundancy level for long-life satellite. MMD of about 30 months

Table 7-3. Baseline EOS-A Orbits

	Thor-Delta Launch Vehicle	Titan Launch Vehicle
Altitude (n mi)	316	387
Days per TM cycle, N	17	17
Revolutions per cycle	254	247
Swathwidth (n mi)	85.2	87.6
Inclination (deg)	100	98.4
Equator crossing time	11 a.m.	11 a.m.

Table 7-4. Orbits for Improved HRPI Coverage

	Thor-Delta Launch Vehicle	Titan Launch Vehicle
Altitude (n mi)	336	376
Days per TM cycle	17	17
Revolutions per cycle	252	248
Days per HRPI cycle		
• 30 deg offset	6	4
• 45 deg offset	4	3

payload (thematic mapper) an option. Also shown is the effect of replacing the TM by a HRPI. Again the four redundancy options of Table 4-1 are considered.

Orbit selection criteria have been discussed in earlier sections. Other than launch vehicle capability, key factors include Shuttle compatibility, ground station visibility, and swathing patterns. In order to assess the feasibility of the two candidate MSS missions (i.e., one on each of two satellites versus two on a single satellite) specific acceptable orbits must be postulated.

Figure 7-2 shows the performance characteristics for two mission variants, four redundancy variants, and four orbits, dealing with single-MSS configurations. Figure 7-3 presents the same data for dual-MSS payloads, with the orbit altitudes constrained by unique swathing considerations when using tandem instruments, as discussed in Section 3.1.2.3. In each figure, check-marks indicate compatible factors, cross-marks indicate incompatibility, and question marks show marginal performance. Note that all performance achievable with a Thor-Delta 3910 can also be attained using a Titan IIIB and spacecraft-integral propulsion.*

Based on these figures, the following conclusions can be drawn:

- Spacecraft visibility from Sioux Falls for all of CONUS can be achieved at all altitudes above about 350 nautical miles (dependent on elevation angle assumed). With a receiving antenna 70 miles southeast of Siox Falls (and a microwave link to Sioux Falls), coverage to as low as 320 nautical miles appears feasible (Figure 3-9).

- At orbit altitudes above 410 nautical miles, the Shuttle with FSS cannot rendezvous. If the FSS is stripped by removing the module exchange equipment SPMS, the retrieval altitude is increased to 440 nautical miles.

- Shuttle launch capability is very altitude dependent (e.g., the tradeoff is 900 pounds of payload per 10 nautical miles of altitude, compared with 20 pounds per 10 nautical miles of altitude for a Thor-Delta 2910). Most altitudes which can be reached for Shuttle servicing can also be achieved by a Shuttle direct launch.

*The converse is not true. Missions marginal with a 3910 will not be marginal with a Titan IIIB.

INSTRUMENT PAYLOAD			OPERATIONAL: 1-MSS				OPERATIONAL, R&D: 1-MSS PLUS TM			
ORBIT FACTORS	REPEAT CYCLE, N (DAYS)		18	17	17	17	18	17	17	17
	REVOLUTIONS PER CYCLE, R		251	240	247	253	251	240	247	253
	ALTITUDE (N MI/KM)		493/913	461/853	387/716	326/603	493/913	461/853	387/716	326/603
	SWATHWIDTH (N MI/KM)		86/160	90/167	88/162	85/158	86/160	90/167	88/162	85/158
DELTA-2910 LAUNCH COMPATIBILITY	ON-ORBIT SERVICEABLE CONFIGURATION	MIN	√	√	√	√	X	X	X	X
		VAR1	√	√	√	√	X	X	X	X
		VAR2	√	√	√	√	X	X	X	X
		NOM	√	√	√	√	X	X	X	X
	NONSERVICEABLE CONFIGURATION	MIN	√	√	√	√	X	?	√	√
		VAR1	√	√	√	√	X	X	?	√
		VAR 2	√	√	√	√	X	X	?	√
		NOM	√	√	√	√	X	X	?	√
DELTA-3910 LAUNCH COMPATIBILITY	ON-ORBIT SERVICEABLE CONFIGURATION	MIN	√	√	√	√	√	√	√	√
		VAR 1	√	√	√	√	√	√	√	√
		VAR 2	√	√	√	√	√	√	√	√
		NOM	√	√	√	√	√	√	√	√
	NONSERVICEABLE CONFIGURATION	MIN	√	√	√	√	√	√	√	√
		VAR1	√	√	√	√	√	√	√	√
		VAR 2	√	√	√	√	√	√	√	√
		NOM	√	√	√	√	√	√	√	√
SHUTTLE COMPATIBILITY	LAUNCH SERVICEABLE CONFIGURATION (DIRECT INJECT)	MIN	X	X	√	√	X	X	√	√
		VAR 1	X	X	√	√	X	X	√	√
		VAR 2	X	X	√	√	X	X	√	√
		NOM	X	X	√	√	X	X	√	√
	LAUNCH NONSERVICEABLE CONFIGURATION (DIRECT INJECT)	MIN	X	X	√	√	X	X	√	√
		VAR 1	X	X	√	√	X	X	√	√
		VAR 2	X	X	√	√	X	X	√	√
		NOM	X	X	√	√	X	X	√	√
	RETRIEVE (NO DEORBIT)		X	X	√	√	X	X	√	√
CONUS COVERAGE FROM SIOUX FALLS			√	√	√	?	√	√	√	?

NOTES:

(1) TITAN III B EXCEEDS 3910 CAPABILITY AND WILL ACCOMMODATE ALL CASES SHOWN IF SPACECRAFT HAS INTEGRAL CIRCULARIZATION PROPULSION

(2) SHUTTLE INDIRECT INJECTION FEASIBLE FOR ALL CASES SHOWN

LEGEND:

MIN = MINIMUM
VAR1 = VARIANT 1
VAR2 = VARIANT 2
NOM = NOMINAL

√ = COMPATIBLE
X = INCOMPATIBLE
? = MARGINAL

Figure 7-2. Operational Single-MSS Missions

INSTRUMENT PAYLOAD			OPERATIONAL: 2-MSS				OPERATIONAL: 2-MSS PLUS TM			
ORBIT FACTORS (1)	REPEAT CYCLE, N (DAYS)		9/17	7/13	7/13	9/17	9/17	7/13	7/13	9/17
	REVOLUTIONS PER CYCLE, R		240	184	193	253	240	184	193	253
	ALTITUDE (N MI/KM)		461/854	454/841	332/615	326/604	461/854	454/841	332/615	326/604
	SWATHWIDTH (N MI/KM)		90/167	118/218	112/208	86/158	90/167	118/218	112/208	86/158
DELTA-2910 LAUNCH COMPATIBILITY	ON-ORBIT SERVICEABLE CONFIGURATION	MIN	X	X	?	?	X	X	X	X
		VAR 1	X	X	?	?	X	X	X	X
		VAR 2	X	X	X	X	X	X	X	X
		NOM	X	X	X	X	X	X	X	X
	NONSERVICEABLE CONFIGURATION	MIN	√	√	√	√	X	X	X	X
		VAR 1	√	√	√	√	X	X	X	X
		VAR 2	√	√	√	√	X	X	X	X
		NOM	√	√	√	√	X	X	X	X
DELTA-3910 LAUNCH COMPATIBILITY (2)	ON-ORBIT SERVICEABLE CONFIGURATION	MIN	√	√	√	√	√	√	√	√
		VAR 1	√	√	√	√	?	?	√	√
		VAR 2	√	√	√	√	?	?	√	√
		NOM	√	√	√	√	?	?	√	√
	NONSERVICEABLE CONFIGURATION	MIN	√	√	√	√	√	√	√	√
		VAR 1	√	√	√	√	√	√	√	√
		VAR 2	√	√	√	√	√	√	√	√
		NOM	√	√	√	√	√	√	√	√
SHUTTLE COMPATIBILITY (3)	LAUNCH SERVICEABLE-CONFIGURATION (DIRECT INJECT)	MIN	X	X	√	√	X	X	√	√
		VAR 1	X	X	√	√	X	X	√	√
		VAR 2	X	X	√	√	X	X	√	√
		NOM	X	X	√	√	X	X	√	√
	LAUNCH NONSERVICEABLE CONFIGURATION (DIRECT INJECT)	MIN	X	X	√	√	X	X	√	√
		VAR 1	X	X	√	√	X	X	√	√
		VAR 2	X	X	√	√	X	X	√	√
		NOM	X	X	√	√	X	X	√	√
	RETRIEVE (NO DEORBIT)		X	X	√	√	X	X	√	√
CONUS COVERAGE FROM SIOUX FALLS			√	√	?	?	√	√	?	?

NOTES:

(1) REPEAT CYCLE VALUES ARE FOR DUAL AND SINGLE INSTRUMENTS OPERATIONAL

(2) TITAN III B EXCEEDS 3910 CAPABILITY AND WILL ACCOMMODATE ALL CASES SHOWN IF SPACECRAFT HAS INTEGRAL CIRCULARIZATION PROPULSION

(3) SHUTTLE INDIRECT INJECTION FEASIBLE FOR ALL CASES SHOWN

LEGEND:

MIN = MINIMUM
VAR1 = VARIANT 1
VAR2 = VARIANT 2
NOM = NOMINAL

√ = COMPATIBLE
X = INCOMPATIBLE
? = MARGINAL

Figure 7-3. Operational Dual - MSS Missions

- In most cases, redundancy is not pivotal in determining launch vehicle (i.e., 2910) applicability. Moreover, since spacecraft cost is not greatly affected by large improvements in MTTF, redundant configuration may be desirable, particularly with a dual-MSS operational payload.

- Serviceability imposes a heavy burden, generally requiring more capability than the Delta 2910 can provide.

For each of the four specific missions of the figures, definitive, but tentative, recommendations can be made:*

- Single MSS, No TM. The 387-nautical mile orbit is selected, providing CONUS coverage and compatibility in all respects, including serviceability. The Delta 2910 launch vehicle should be used.

- Single MSS, TM. The 326-nautical mile orbit with an unserviceable, 2910 launched configuration is suggested. This can be made resuppliable if launched with a 3910 (or equivalent).

- Dual MSS, No TM. The 326-nautical mile orbit is recommended, giving a good swathwidth, 9-day coverage with two MSS instruments active, and 17-day coverage with one active. The unserviceable configuration, 2910 launched, is suggested.

- Dual MSS, TM. The 326-nautical mile orbit is recommended, as above. The 3910 or equivalent (e.g., Titan IIIB) is required. Serviceability will not penalize launch vehicle selection and so should be provided.

These recommendations, particularly the constraints based on Shuttle retrieval capability, are heavily flavored by retrieval/resupply considerations. Launches prior to the Shuttle era must have redundancy to make retrieval likely, in order to make these recommendations meaningful. As stressed above, final recommendations must await mission cycle-Shuttle applicability studies to be treated in Report 6.

---

*Final decision must be based on mission cycle studies, considering Shuttle costs, etc.